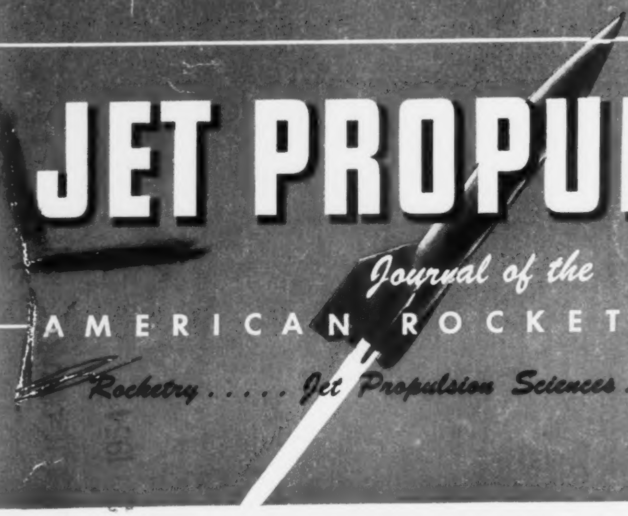
VOLUME 24 JANUARY-FEBRUARY 1954 NUMBER 1



JET PROPULSION

Journal of the
AMERICAN ROCKET SOCIETY

Rocketry . . . . Jet Propulsion Sciences . . . . Astronautics

VOLUME 24 JANUARY-FEBRUARY 1954 NUMBER 1

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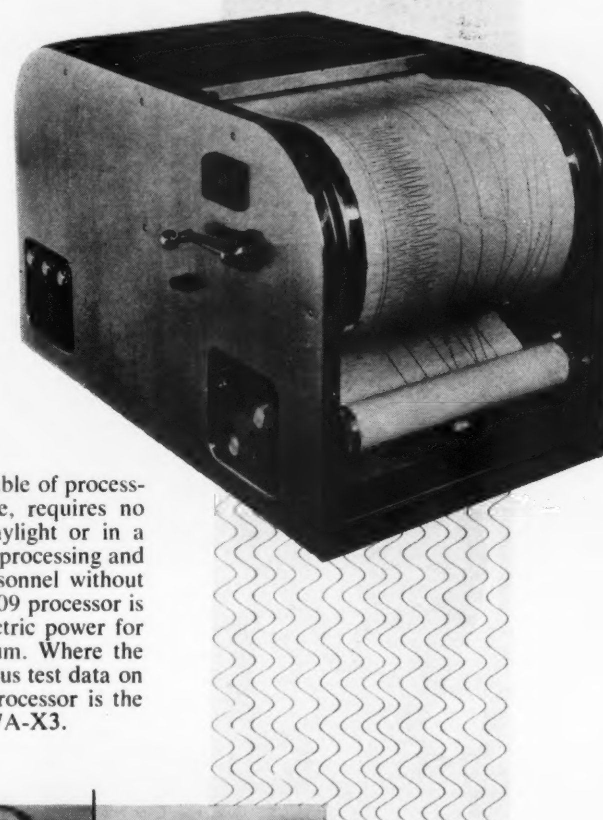

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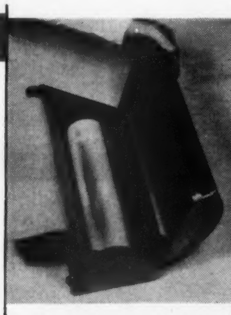

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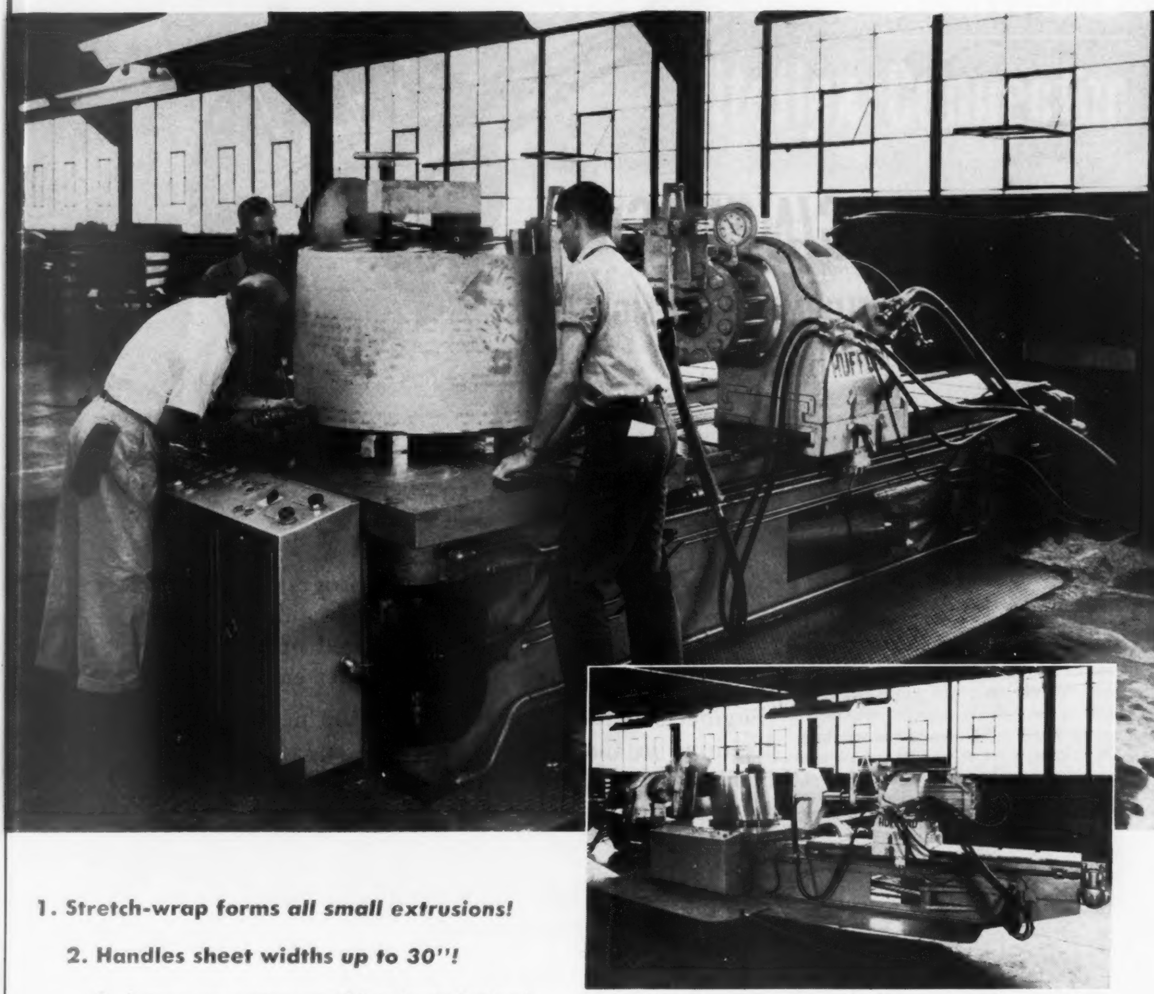

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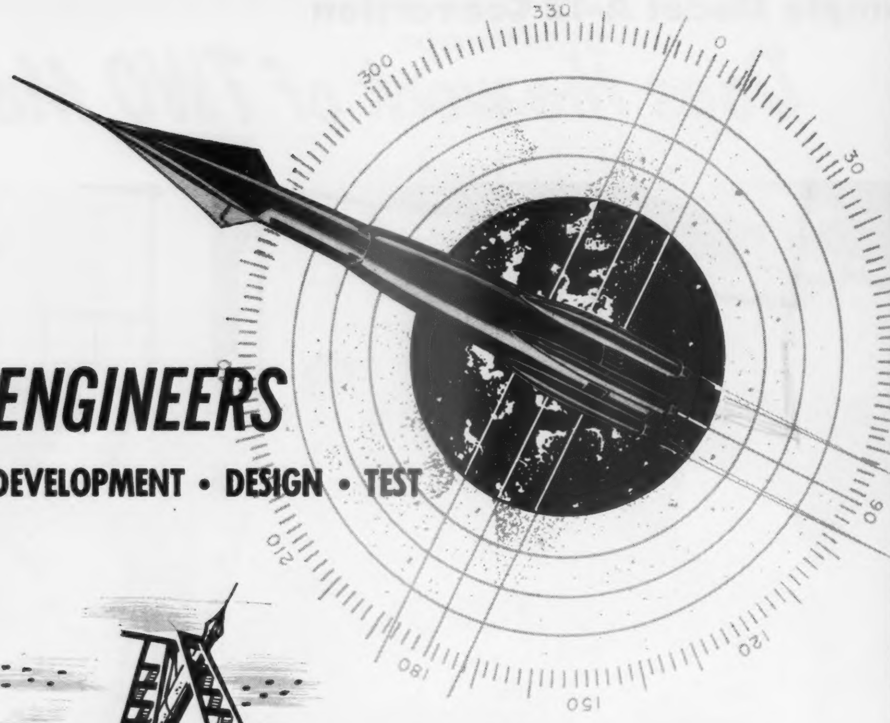

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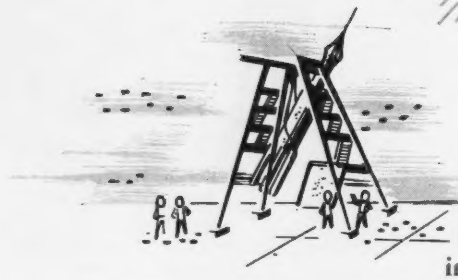

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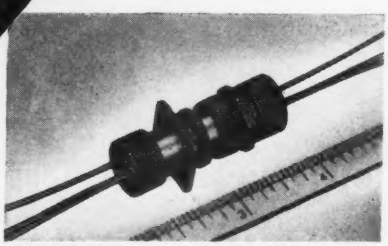

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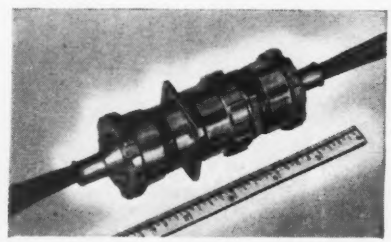

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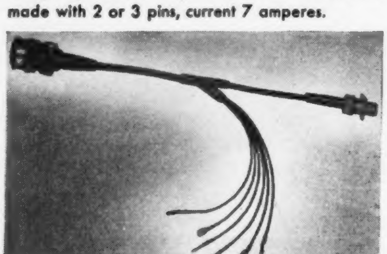

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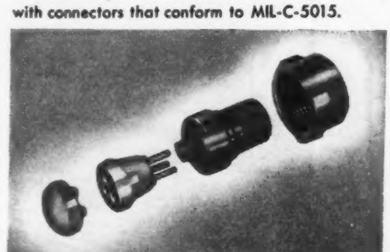

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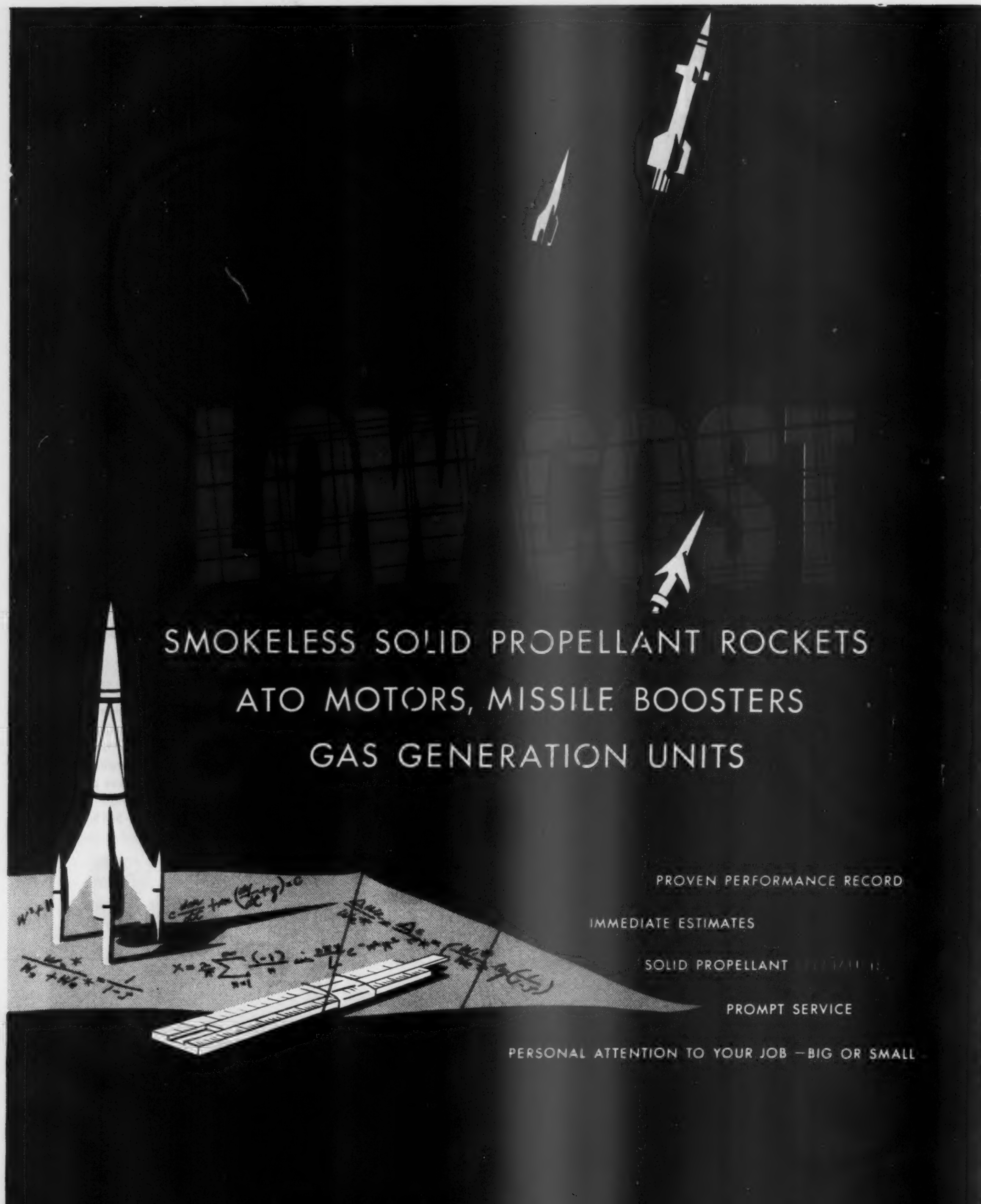
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Editor-in-Chief

MARTIN SUMMERFIELD

1954-1955

Sci R v. 24-25

Erosive Burning of Some Composite Solid Propellants

LEON GREEN, JR.[1]

Aerojet-General Corporation, Azusa, Calif.

Experimental data on the erosive burning of several omposite solid propellants are reported and discussed. he data appear to be best correlated in terms of a reduced ass velocity. At low Mach numbers, this correlation ecomes equivalent to several widely used forms employing near velocity. It is found that the erosive effect is greater n slow-burning propellants than in fast-burning ones, an bservation consistent with previously reported data for rious double-base propellants. The existence of an aprent "threshold velocity for erosion" is tentatively atributed to the failure of the one-dimensional idealization o represent the true flow conditions obtaining at the fore nd of the grain, but it is emphasized that more compreensive studies of the erosive-burning phenomenon will e required in order to resolve the threshold velocity quesion.

SERIAL

Nomenclature

= cross-sectional of flow channel or grain perforation. Usually called port area (in.^{2})
= cross-sectional area of nozzle throat (in.^{2})
= speed of sound corresponding to T_0 (ft/sec)
= specific gas constant, R/μ
= constant in zero-velocity burning rate equation $r_0 = cp^n$, (in./sec) (lbf/in.^{2})$^{-n}$
= mass flow (or discharge) coefficient of propellant (sec/ft). If mass is measured in lbm, C_w is expressed in the units lbm/lbf sec. Theoretically, $C_w = \Gamma(bT_0)^{-1/2}$
= cross-sectional-average mass velocity in flow channel (lbm/in.^{2} sec)
= critical value of G required to produce $M = 1$ in constant-area channel (lbm/in.^{2} sec)
= A_t/A = ratio of nozzle throat area to port area at aft end of grain
= ratio of burning surface area to nozzle throat area
= erosive-burning constant in terms of reduced mass velocity
= erosive-burning constant in terms of reduced linear velocity
= erosive-burning constant in terms of throat-to-port area ratio
= erosive-burning constant in terms of mass velocity (in.^{2} sec/lbm)
= erosive-burning constant in terms of linear velocity (sec/ft)
= length of propellant grain (in.)
= Mach number of gas flow in grain perforation
= exponent in zero-velocity burning rate law, $r_0 = cp^n$

p = static pressure (lbf/in.^{2})
p_0 = static pressure at $x = 0$ (fore-end condition) (lbf/in.^{2})
p_s = isentropic stagnation pressure (lbf/in.^{2})

$$R = \text{gas constant} = 10.73 \left(\frac{\text{lbf}}{\text{in.}^2}\right)\left(\frac{\text{ft}^3}{\text{lbm mol}}\right)\left(\frac{1}{\text{deg R}}\right)$$

$$= 49{,}800 \left(\frac{\text{slug}}{\text{ft sec}^2}\right)\left(\frac{\text{ft}^3}{\text{slug mol}}\right)\left(\frac{1}{\text{deg R}}\right)$$

r = burning rate of propellant under given conditions of pressure, propellant temperature, and gas velocity (in./sec)
r_0 = burning rate of propellant under same conditions of pressure and propellant temperature, but with no gas velocity (in./sec)
T = gas temperature (deg F or R)
T_0 = isobaric flame temperature of propellant (stagnation temperature of gas flow) (deg F or R)
v = cross-sectional-average linear gas velocity in grain perforation (ft/sec)
v_0 = threshold velocity for erosion (ft/sec)
x = distance from fore end of grain to point in question (in.)
γ = ratio of specific heats of propellant gases

$$\Gamma = \sqrt{\gamma}\left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}}$$

ϵ = erosion ratio, r/r_0
μ = molecular weight of propellant gases
ρ = mass density of propellant gases
ρ_π = mass density of solid propellant

Introduction

THE term "erosive burning" is used to indicate the dependence of the burning rate of a solid propellant upon the velocity of gas flow parallel to the burning surface. Although its influence can now be recognized in the early experiments of Mansell (1),[2] this phenomenon was apparently first identified by Muraour (2). More recently, its importance in the design of solid propellant rockets has been described by Wimpress and others (3, 4). A theoretical study of the phenomenon has been presented by Corner (5), but the simplifying assumptions involved therein render the theory semiquantitative at best. For practical purposes of propellant charge design, it is generally assumed that the effects of gas velocity upon burning rate can be correlated in terms of an "erosion ratio," ϵ, expressed as a function of velocity

$$\epsilon \equiv \frac{r}{r_0} = f\,(\text{velocity}) \quad [1]$$

Several different analytical approximations for the dependence in question have been advanced in the literature. The ex-

Presented at the 8th ARS National Convention, New York, .Y., December 3, 1953

[1] Senior Engineer, Solid Engine Department. Member ARS.

[2] Numbers in parentheses refer to References on page 15.

AUG 2 1957

pression first proposed (on the basis of British experiments with double-base propellants) relates the erosion ratio to the linear gas velocity

$$\epsilon = 1 + k_v v \qquad [2]$$

This expression is supported by subsequent investigations, notably those of Thompson and McClure (6) and Touchard (7). A frequently used dimensionless erosion constant is defined in terms of a reduced linear velocity by the equivalent expression

$$\epsilon = 1 + k_1 \frac{v}{a_0} \qquad [3]$$

Some experiments have suggested the existence of a "threshold velocity" for erosion. Such an effect can be expressed as

$$\epsilon = 1 + k_v(v - v_0)\, 1(v - v_0) \qquad [4]$$

where $1(v-v_0)$ is a unit step function, equal to zero when $v < v_0$ and equal to unity when $v \geq v_0$. The possibility that such a threshold effect (also indicated by the present experiments) is apparent and not real is considered in the following discussion.

On the basis of earlier studies at this laboratory (8), an approximation expressed in terms of mass velocity was suggested

$$\epsilon = 1 + k_G G \qquad [5]$$

These experiments were performed at approximately the same chamber-pressure level, however, and the data obtained could have been correlated equally well in terms of linear velocity. Mass velocity apparently was chosen on the basis that it is the physically pertinent variable in the more familiar problem of convective heat transfer in uniform flow along a solid boundary. Another correlation of this type has been reported subsequently (9) but appears questionable on the basis of factors considered in the following discussion. The data obtained during the present study cannot be correlated by Equation [5].

Estimates of the erosion effect are often made on the basis of the approximation

$$\epsilon \cong 1 + k_2\left(\frac{x}{L}\right)\left(\frac{A_t}{A}\right) \qquad [6]$$

This expression is based upon Equation [2] and involves the assumptions that the gas velocity in the constant-area channel varies linearly with x and that the pressure drop along the grain is small. For these reasons k_2 is not a true constant. Despite its limitations, Equation [6] is often useful for preliminary design purposes.

Equipment and Procedure

No practical method of determining the instantaneous burning rate of a solid propellant grain is now known. Accordingly, the present study employed the standard partial-burning technique, in which the propellant grain is extinguished after burning for a recorded interval of time and the amount of propellant burned off at various points along the grain is measured. The gas pressure and velocity at these points are than calculated using the time-average dimensions of the flow channel. In order for the time-average values of burning rate, velocity, and pressure yielded by this technique to approximate the desired instantaneous values with sufficient accuracy, the burning time must be short. On the other hand, unless burning is allowed to progress for an easily measured distance, random errors in grain measurements may be prohibitively large. In addition, short-duration tests magnify errors in time measurements, and the tacit assumption that all parts of the burning surface ignite simultaneously becomes questionable. It may be seen from these considerations that determinations of erosive-burning properties by the partial-burning technique are necessarily approximate in nature. During some of the present tests the cross-sectional area of the flow channel increased significantly. For this reason, use of the geometric time-average area was required in the computations (10). In the most extreme cases of area change during burning (tests of Propellant 2) the diameter of the grain perforation almost doubled, and comparison of the mass velocities computed from both the arithmetic and geometric mean values of A revealed differences approaching 25 per cent. In such an extreme case the use of an average area introduces an error of uncertain magnitude, and the validity of the result is open to question.

The test chamber used for the interrupted-burning experiments is shown in cross section in Fig. 1.[3] The cover plate

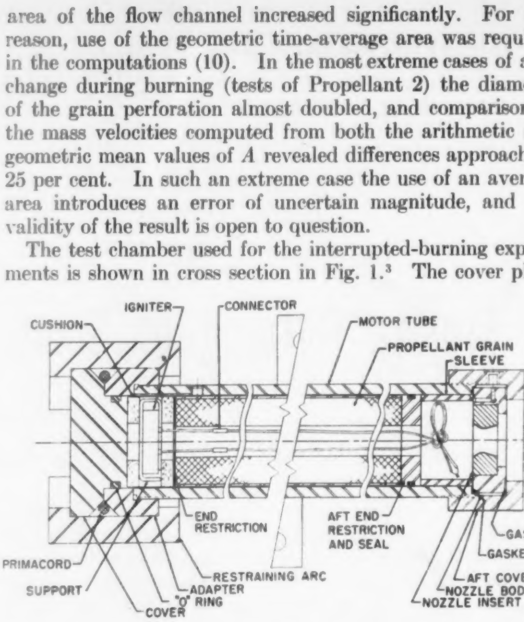

FIG. 1 CROSS-SECTIONAL VIEW OF TEST CHAMBER FOR INTERRUPTED-BURNING EXPERIMENTS

sealing the fore end of the chamber is secured by two restraining arcs. Prior to firing, the arcs are held in place by cloth tape; during firing they are kept in place primarily by friction at the lip. A 12-in. length of Primacord (0.25-in. nominal diameter) is wrapped in the groove under the restraining arcs. In operation, a blasting cap, set off electrically at a predetermined time, is used to ignite the Primacord which in turn blows off the restraining arcs. The chamber pressure then ejects the fore cover, and the pressure falls. The grain, usually extinguished by this rapid decompression and its concurrent cooling action, is ejected from the chamber (fore end pointed downward) into a tank of water which prevents re-ignition. No destruction of metal parts is required by the operation described above; the restraining arcs strike a cylindrical guard ring surrounding the fore end of the motor, drop into the water, and are re-usable.

The propellant grains used in the interrupted-burning tests were tubular, with an outer diameter of 2.5 in., an inner diameter of about 1.0 in., and a length of about 26 in.

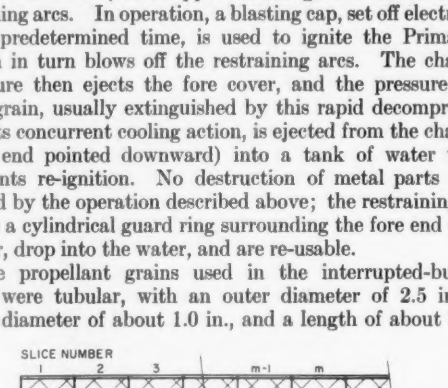

FIG. 2 SCHEMATIC CROSS SECTION OF A PARTIALLY BURNED TUBULAR GRAIN

The ends and outer periphery of the grains were restricted, so that burning occurred only on the inner surface. In order to preserve an approximately uniform port area during the test

[3] This figure shows the heavy restriction and O-ring seal (preventing gas flow along the outer surface of the grain) located at the aft end of the grain. In the present studies, however, this seal was employed on the fore end in order to minimize any constriction of the gas flow leaving the grain; the aft end was restricted by a thin, erodible disk.

(to compensate for the action of erosive burning), the internal channel was given a slight (0.8 per cent) taper, being smaller at the aft end. Following the test firing, the partially burned grains were cut into sections 1 in. thick, as indicated in Fig. 2. Measurements of the internal diameter of the fore and aft end of each section were made at two positions 90 deg apart. Since the fore end of one section corresponds to the aft end of the preceding section, the diameter recorded at each location along the grain length (except for the two extremities) represented an average of four independent measurements. The average mass velocity at the different points along the grain was computed by the mass-balance (neglecting the density of the gas in comparison with that of the solid propellant) for one-dimensional flow

$$G = \frac{\rho_\pi}{A}\int_0^x rZ\,dx \qquad [7]$$

In the stepwise integration employed, this relation was approximated as

$$G(x_m) = \frac{\rho_\pi}{A(x_m)}\sum_1^m \bar{r}(x_{m-1/2})\,\bar{Z}(x_{m-1/2})\Delta x' \qquad [8]$$

where the computation procedure is outlined in Table 1.

TABLE 1 OUTLINE OF MASS FLOW CALCULATION

x_m	= distance from fore end of grain to aft end of slice number m
$D_i(x_m)$	= initial diameter of channel at position x_m
$D_f(x_m)$	= final diameter of channel (after burning-time interval Δt) at position x_m
$D(x_m)$	= $^1/_2[D_i(x_m) + D_f(x_m)]$ = arithmetic time-average channel diameter at position x_m
$\bar{D}(x_{m-1/2})$	= $^1/_2[D(x_m) + D(x_{m-1})]$ = space mean of time-average diameters at positions x_m and x_{m-1}
$A(x_m)$	= $\frac{\pi}{4}D_i(x_m)D_f(x_m)$ = geometric time-average port area at position x_m
$\bar{Z}(x_{m-1/2})$	= $\pi\bar{D}(x_{m-1}/2)$ = space mean of time-average perimeters at positions x_m and x_{m-1}
$r(x_m)$	= $\frac{1}{2\Delta t}[D_f(x_m) - D_i(x_m)]$ = time-average burning rate at position x_m
$\bar{r}(x_{m-1/2})$	= $^1/_2[r(x_m) + r(x_{m-1})]$ = space mean of time-average burning rates at position x_m and x_{m-1}

The pressure, temperature, and velocity distributions along the grains were calculated by considering the true conditions obtaining in the burning grain perforation to be idealized by the assumption of steady, frictionless, one-dimensional, adiabatic flow of a perfect gas in a constant-area channel with mass addition normal to the flow direction, in the fashion of Shapiro and Hawthorne (11). Several relationships for such flows[4] which are pertinent to internal-ballistic calculations are shown in Fig. 3 as functions of Mach number for the case of $\gamma = 1.26$. These relationships are more familiar and more useful to grain designers when plotted as functions of G/G^* (Fig. 4), the experimentally determined variable in erosive-burning studies.

If the ratios of p/p_0 and T/T_0 given in Fig. 3 are written in terms of mass velocity instead of Mach number and are combined (with use of the perfect gas law), the pressure ratio may be expressed as

$$\frac{p}{p_0} = \frac{1}{\gamma+1} + \frac{\gamma}{\gamma+1}\left[1 - 2\left(\frac{\gamma+1}{\gamma}\right)\frac{bT_0G^2}{p_0^2}\right]^{1/2} \qquad [9]$$

From this result it may be seen that the critical mass velocity G^* required to produce Mach number unity in the constant-area channel is

$$G^* = \left[\frac{\gamma}{\gamma+1}\,\frac{p_0^2}{2bT_0}\right]^{1/2} \qquad [10]$$

and is a function of the propellant composition and the fore-end chamber pressure (10). With the time-average pressure p_0 known from the experimental pressure vs. time curve and G determined from measurements of the partially burned grain, the pressure, velocity, and temperature in the gas flow at any

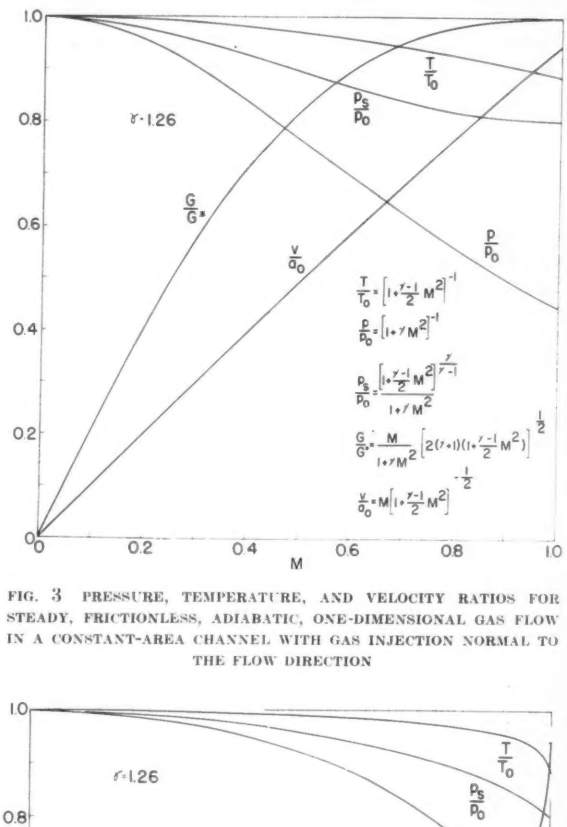

FIG. 3 PRESSURE, TEMPERATURE, AND VELOCITY RATIOS FOR STEADY, FRICTIONLESS, ADIABATIC, ONE-DIMENSIONAL GAS FLOW IN A CONSTANT-AREA CHANNEL WITH GAS INJECTION NORMAL TO THE FLOW DIRECTION

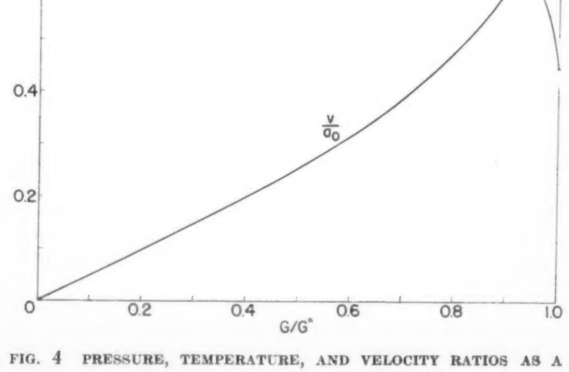

FIG. 4 PRESSURE, TEMPERATURE, AND VELOCITY RATIOS AS A FUNCTION OF REDUCED MASS VELOCITY

[4] Fig. 3 does not completely illustrate the "choking" effect which limits the velocity attainable by mass addition to a constant-stagnation-temperature flow of gas in a constant-area channel. As shown by Shapiro and Hawthorne, mass addition to a supersonic stream will also cause the Mach number to approach unity. Mass addition to a flow at $M = 1$ will not increase M, but only causes the boundary conditions to change, i.e., the pressure level increases. This effect determines the maximum internal area ratio (burning surface area to port area) usable in a solid propellant rocket motor intended to operate over a given pressure range.

point along the grain can be determined by the use of Equation [12] and Fig. 4, or its equivalent for the value of γ in question.

Experimental Results

The propellants used in this study were of a heterogeneous nature, consisting of inorganic oxidizer particles cast in a matrix of organic fuel. Pertinent thermodynamic properties of the combustion products of these propellants are presented in Table 2. The experimental curves of chamber pressure

TABLE 2 APPROXIMATE THERMODYNAMIC PROPERTIES OF THE COMBUSTION PRODUCTS OF SOME COMPOSITE SOLID PROPELLANTS

Propellant number	Specific heat ratio, γ	Flame temperature T_0, °F	Molecular weight, μ
1	1.26	3700	24.5
2	1.26	3640	23.4
3	1.26	3540	24.0
4	1.27	3510	22.9
5	1.25	3800	22.9
6	1.21	4800	27.2
7	1.27	2940	22.1
8	1.30	2090	19.5

vs. time obtained in three interrupted-burning tests of Propellant 1 are presented in Fig. 5. These curves are typical of the pressure-time records obtained with an internal-burning, tubular charge of a propellant when the erosive effect is not pronounced (i.e., when the throat-to-port area ratio J is relatively small). The experimental burning-rate data ob-

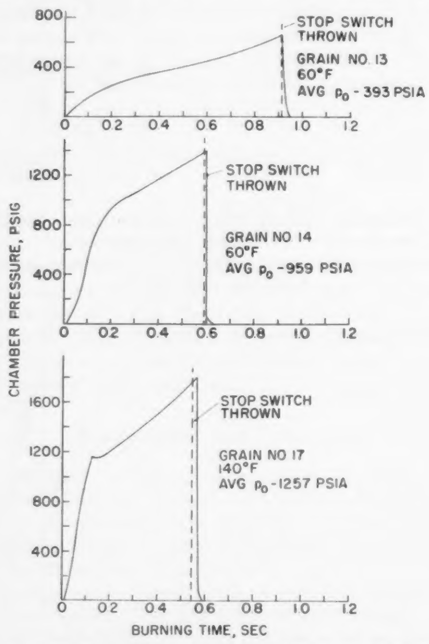

FIG. 5 PRESSURE-VS.-TIME CURVES OBTAINED IN INTERRUPTED-BURNING TESTS OF PROPELLANT 1

tained in these tests are presented in Fig. 6. These curves are typical of those obtained with the other propellants. The mass flow rates observed in the same tests are presented in Fig. 7 as a function of the distance from the fore end of the grain; the nonlinearity observed there manifests the erosive-burning effect.

Correlation of the experimental data obtained during the present study by the different expressions mentioned in the introduction was investigated. For this purpose the velocity and pressure at each position along the grain were computed

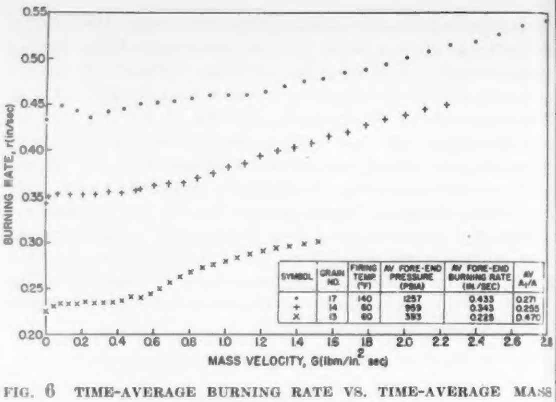

FIG. 6 TIME-AVERAGE BURNING RATE VS. TIME-AVERAGE MASS VELOCITY (PROPELLANT 1)

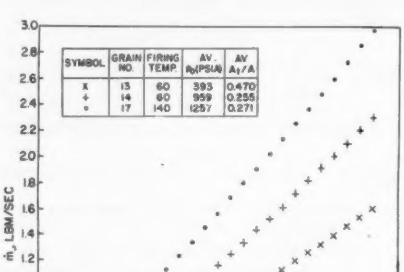

FIG. 7 MASS FLOW RATE VS. DISTANCE FROM FORE END OF GRAIN (PROPELLANT 1)

by the method outlined above, and the zero-velocity burning rate r_0 at each position was determined from the fore-end burning rate by a correction for the change due to the pressure drop along the grain.[5] This correction generally may not be ruled out a priori, and it is especially important when the throat-to-port area ratio J is greater than about 0.5. When the change in zero-velocity burning rate along the grain amounts to only a few per cent, it may validly be neglected for many purposes. In determinations of erosive-burning constants, however, such a procedure is not permissible, since a small error in the quantity $(1 + \eta)$, where η is small compared to unity, may reflect a significant error in η.

The data of Fig. 6 are presented in Fig. 8 in terms of ϵ vs. G. It is evident that the data are not correlated in terms of mass velocity;[6] it was found that in order to compensate for the differences in pressure level prevailing during the different tests, a correlation in terms of either linear velocity or reduced mass velocity would be required. The data on other propellants studied showed this same behavior. In this connection, it was noted that the erosive-burning data for JPN ballistite and a German double-base propellant presented

[5] The zero-velocity burning rates of the propellants under investigation show the usual pressure dependence, $r_0 = cp^n$.

[6] As mentioned earlier, a correlation of the type $\epsilon = 1 + k_G G$ has been reported at another laboratory (9), but in view of two sources of possible error in the experimental procedure, this correlation must be regarded as tentative. The first source is the absence of a correction for the effect of pressure drop along the grain, as was described above. An additional uncertainty arises from the use of average values as approximations of instantaneous values, since the data were obtained in experiments where the cross-sectional port area was allowed to increase during the interval of burning by factors ranging up to 10.

by Wimpress (3) deviated markedly from the linear expressions of Equations [2], [3], or [4], showing at high velocities a curvature concave downward when plotted in terms of ϵ vs. v. The data for JPN, for instance, are shown in Fig. 9. The similar curvature exhibited by the one-dimensional gas-dynamic relationship between reduced mass velocity and Mach number, as shown in Fig. 3, suggested a possible correlation in terms of reduced mass velocity

$$\epsilon = 1 + k\,\frac{G}{G^*} \qquad [11]$$

The data of Fig. 6 in terms of ϵ vs. G/G^* are presented in Fig. 10. It may be noted that the data appear to be best fitted by a "threshold" approximation

$$\epsilon = 1 + k\left[\frac{G}{G^*} - \left(\frac{G}{G^*}\right)_0\right] 1 \left[\frac{G}{G^*} - \left(\frac{G}{G^*}\right)_0\right] \qquad [12]$$

Accordingly, the erosive-burning constants cited herein

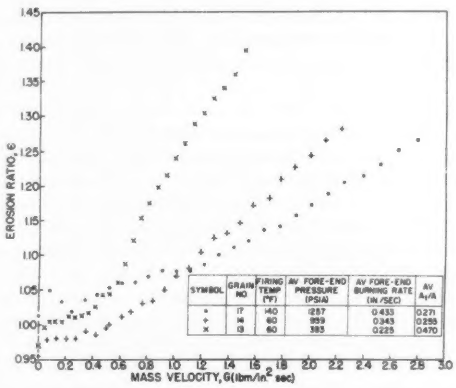

FIG. 8 EROSION RATIO VS. MASS VELOCITY (PROPELLANT 1)

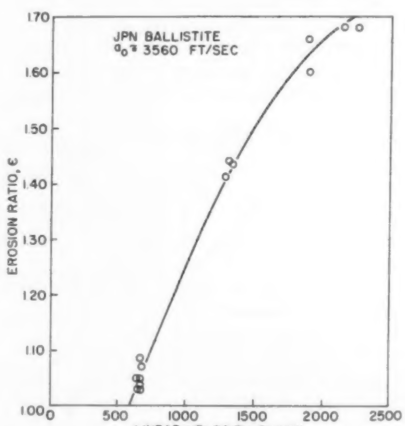

FIG. 9 EROSION RATIO VS. LINEAR VELOCITY (JPN BALLISTITE)

represent values of the slope k in Equation [12]; Fig. 10 yields an average value of $k = 0.80$. Since the threshold phenomenon is considered to be merely an apparent effect (as discussed later), however, it is assumed that for most engineering estimates the k-values defined by Equation [12] can be employed in the simpler Equation [11]. Such an identification is rationalized on the ground that a grain design procedure which neglects the apparent threshold effect[7] in this manner leads to a conservative estimate of the chamber pressure. Since at low Mach numbers the flow relationships

[7] The partial-burning experiments conducted during the present study yielded rough $(G/G^*)_0$ values of the order 0.1–0.2.

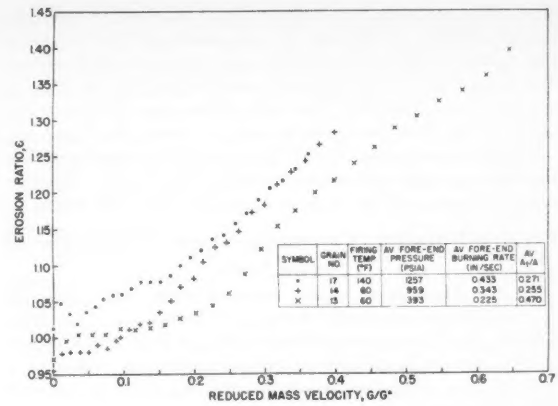

FIG. 10 EROSION RATIO VS. REDUCED MASS VELOCITY (PROPELLANT 1)

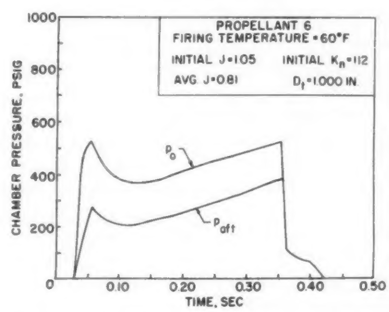

FIG. 11 PRESSURE-VS.-TIME CURVES OBTAINED IN INTERRUPTED-BURNING TEST OF PROPELLANT 6

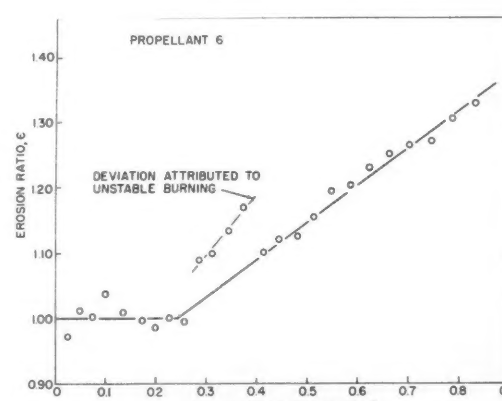

FIG. 12 EROSION RATIO VS. REDUCED MASS VELOCITY (PROPELLANT 6)

of Fig. 3 reduce to $G/G^* \cong M[2(\gamma + 1)]^{1/2}$ and $v/a_0 \cong M$, the proposed correlations of Equations [2], [3], and [6] become equivalent to Equation [11], and the various erosive burning constants are related as follows

$$k_v = k_1/a_0 \qquad [13]$$

$$k_1 \cong k[2(\gamma + 1)]^{1/2} \qquad [14]$$

$$k_2 \cong k_v b T_0\, C_w \cong k_1 \left(\frac{2}{\gamma + 1}\right)^{\frac{\gamma + 1}{2(\gamma - 1)}} \qquad [15]$$

Only one of the tests performed in the present brief investigation employed a throat-to-port area ratio high enough to permit a choice between the correlation employing linear velocity and that using reduced mass velocity. The pressure-

time curve (with its high erosion peak) obtained in this test is shown in Fig. 11 and may be contrasted with the curves of Fig. 5, which were obtained with more conservative J-values. The erosion ratios obtained are presented vs. G/G^* in Fig. 12 and vs. v/a_0 in Fig. 13. The latter figure supports the trend observed by Wimpress; tentatively, at least, the linear approximation in terms of G/G^* (Equation [11]) is preferred for the design of high-performance motors with $J \cong 1$. Further work, however, will be required to substantiate this belief.

Erosive-Burning Constants from Conventional Static Tests

In the absence of direct measurements from partially burned grains, an erosive-burning constant for two slow-burning propellants was derived indirectly from conventional static tests (with motors of various sizes) by means of a widely used design equation for the equilibrium fore-end chamber pressure (4, 6)

$$p_0 \cong \left\{\frac{c(\rho_\pi - \rho)K_n[1 - {}^1/_3\Gamma^2J^2]^n[1 + {}^1/_2k_2J]}{C_w[1 - {}^1/_2\Gamma^2J^2]}\right\}^{1/(1-n)} \quad [16]$$

This expression is based upon a surface-average burning rate approximated by letting $x/L = 1/2$ in Equation [6] and by using a space-average chamber pressure in the zero-velocity burning-rate law. The denominator is, in effect, a "modified discharge coefficient" based upon the fore-end pressure, rather than upon the isentropic stagnation pressure at the nozzle entrance; the factor $[1 - {}^1/_2\Gamma^2J^2]$ provides a first-order correction for the drop in stagnation pressure along the grain. Since slow-burning propellants exhibit measurable erosive effects at moderate Mach numbers (conservative J-ratios), inversion of this equation for determination of k_2 from knowledge of p_0 is permissible. Neglecting ρ in comparison with ρ_π, the expression for k_2 is

$$k_2 \cong \frac{2}{J}\left\{\frac{C_w[1 - {}^1/_2\Gamma^2J^2]p_0^{1-n}}{c\rho_\pi K_n[1 - {}^1/_3\Gamma^2J^2]^n} - 1\right\} \quad [17]$$

Significant erosive effect is required for use of this equation, otherwise the first term approaches unity and the error involved may be prohibitive. For this reason, Equation [17] is not well suited for use with fast-burning propellants, since large J-ratios reduce the accuracy of the pressure drop correction.

The erosive-burning constants for the several propellants studied during the present investigation, constants obtained either by direct grain measurements or by ballistic tests, are summarized in Table 3. It may be noted that either method

TABLE 3 EROSIVE-BURNING CONSTANTS FOR SOME COMPOSITE SOLID PROPELLANTS

Propellant number	Number of tests	Average erosive-burning constant, k	Burning rate at standard conditions,* (in./sec)
Obtained by partial-burning technique			
1	3	0.80	0.37
2	5	0.78	0.32
3	1	0.83	0.36
4	1	0.85	0.28
5	2	1.3	0.21
6	1	0.58	0.36
Obtained by conventional static tests			
7	4	2.6	0.084
8	4	6.1	0.060

* Standard conditions: p = 1000 psia, T = 60 F, negligible gas velocity parallel to surface.

(Equation [11] or [17]) involves the determination of a small difference between two relatively large quantities. Accordingly, it must be understood that the results are approximate.

Discussion

From the data given in Table 3 and graphically represented in Fig. 14, it is evident that the erosive-burning effect is more

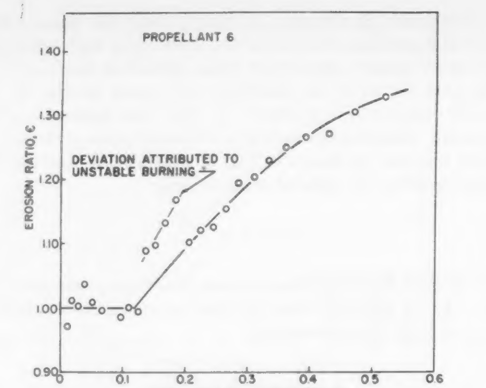

FIG. 13 EROSION RATIO VS. REDUCED LINEAR VELOCITY (PROPELLANT 6)

pronounced in slow-burning propellants than in fast-burning ones. This dependence has long been known (3); a similar general trend is shown by the points for double-base propellants, which are rough, over-all averages of the data of Thompson and McClure (6) and are included merely for comparison. (It should be noted that the standard-condition burning rate for these propellants is based upon a reference temperature of 77 F instead of 60 F.) It is generally believed that the erosive-burning effect is attributable to an enhanced rate of heat transfer to the solid propellant attending the presence of a gas flow parallel to the burning surface. Qualitatively, this dependence may be envisaged as a boundary-layer effect; the greater the velocity, the thinner the boundary layer and the steeper the temperature gradient from the hot core to the cooler solid surface. From this point of view, a fast-burning propellant would be expected to exhibit a relatively small erosion effect, since the rapid evolution of gas normal to the burning surface would produce a thicker boundary layer (for a given parallel velocity) than would be obtained in the case of a slow-burning propellant. A quantitative boundary-layer approach to the erosive-burning problem will be complicated, however, by the strong pressure and temperature gradients involved, and will be further impeded

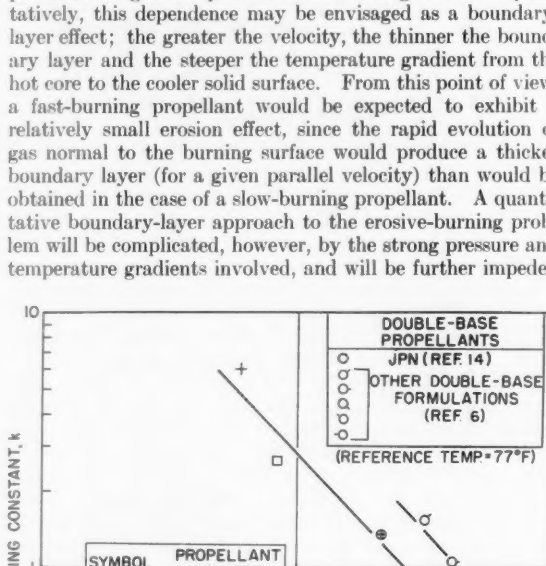

FIG. 14 AVERAGE EROSIVE-BURNING CONSTANT (IN TERMS OF REDUCED MASS VELOCITY) OF SEVERAL SOLID PROPELLANTS VS. NOMINAL BURNING RATE OF PROPELLANT UNDER STANDARD CONDITIONS

by the rudimentary understanding of the combustion process prevailing at present. In the absence of any such treatment permitting a theoretical prediction of the erosive-burning characteristics of a new propellant formulation, the crude, purely empirical correlation suggested in Fig. 14 can provide rough estimates of the erosion constants (of propellants of the type under discussion) for preliminary design purposes.

Perhaps the most interesting feature of the various erosion ratio curves presented herein is that they all suggest the existence of a threshold velocity for erosion, as mentioned earlier. It is believed that in the present experiments, at least, this effect is not real but apparent, that it may be attributed to the failure of the gross, one-dimensional approximation to represent the true flow conditions obtaining at the fore end of the grain perforation. The idealization of one-dimensional flow assumes that the actual velocity at any point in the channel may be satisfactorily approximated by a cross-sectional average velocity. Such an approximation obviously breaks down at the fore end of the channel and becomes accurate only after the flow has progressed a certain distance downstream, as depicted qualitatively in Fig. 15; the cross-sectional average mass velocity computed from Equation [7] will be considerably higher than the true velocity prevailing

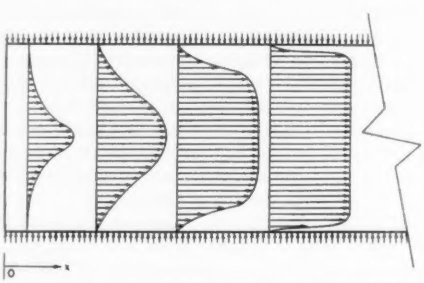

FIG. 15 HYPOTHETICAL VELOCITY DISTRIBUTION NEAR THE FORE END OF GRAIN PERFORATION

near the boundary. Since the magnitude of the observed erosion ratio is governed by the true flow conditions, near the fore end of the channel it will necessarily be smaller than that predicted on the basis of a cross-sectional average quantity. The existence of the apparent threshold in the present experiments is therefore attributed to the fact that a finite "transition distance" is required for the establishment of a flat velocity profile and a well-defined boundary layer. This surmise is supported by the fact that earlier erosive-burning studies at this laboratory (8), using solid cylindrical grains burning on the fore end as well as on the lateral periphery (thus producing a relatively flat velocity profile at the channel entrance) revealed no such pronounced threshold. On the other hand, Wimpress's measurements (Fig. 9), which were made upon small cylindrical samples burning in a stream of gas produced by a separate grain, indicate a threshold effect. In view of the rather crude nature of the interrupted-burning experiments conducted to date, it is believed that the threshold-velocity question will be resolved only by further studies performed with a refined experimental technique. For the time being, the simple expressions of Equations [2], [3], and [11] are satisfactory for the purposes of grain design.

It may be noted that four of the points of Figs. 12 and 13 deviate abnormally from the general trend, indicating a small region exhibiting an exaggerated burning rate. This deviation is attributed to the presence of localized unstable or "resonant" burning, a phenomenon which occurs frequently in tests of high-energy propellants (3). The propellant in question consistently exhibits pronounced unstable burning in tests at −40 F. Recent experiments by Smith and Sprenger (13), using propellant grains of the same type employed during the present study, have established the fact that unstable burning is inseparably associated with the presence of high-amplitude acoustical oscillations of the gases in the burning cavity. Both axial ("organ-pipe") and tangential mobes were identified, and it was found that the most pronounced burning irregularities were associated with the tangential waves. Owing to the localized nature of the "resonance" in the test of Fig. 11 (conducted at 60 F, at which temperature the instability encountered with Propellant 6 is mild and infrequent), the pressure-time curve does not exhibit the so-called "secondary pressure peaks" which characterize a highly unstable test.

No significant variation of erosive-burning properties with testing temperature was noted in any of the previous investigations mentioned above. In the present experiments, accordingly, no systematic study was attempted; most of the tests were conducted at 60 F. In some cases the temperature was varied, but no significant effect upon the erosion constant was observed. In view of the trend shown in Fig. 14 and of the temperature-dependent burning rates of solid propellants, however, it is suggested that more refined erosive-burning experiments may reveal some dependence of the erosion constant upon testing temperature.

Conclusion

The foregoing discussion has described a brief experimental study of the erosive burning of some composite solid propellants, a study encompassing a range of propellant burning rates somewhat greater than any covered in previous investigations. It was suggested that the erosion ratios observed might better be correlated in terms of a reduced mass velocity than by linear velocity, but, owing to the limited scope of the investigation, the superiority of this proposed correlation was not conclusively established. The desirability of further study must be emphasized; in order to determine the best method of correlation and to resolve the question of the threshold velocity, experiments of increased precision are necessary. X-ray movies of rocket motors during firing have been under discussion as a method of determining burning rates for at least a decade, but the author is unaware of any data obtained by such a technique. High-speed photographic recording of short-duration, interrupted-burning tests in a two-dimensional, transparent-walled chamber might prove more practical.

Acknowledgments

The author wishes to thank the many members of the Aerojet organization who contributed to this program. Special acknowledgment is made to R. P. Smith, who designed the interrupted-burning chamber, initiated the experimental work and, together with D. F. Sprenger, provided helpful suggestions during the course of the program. Propellant processing and motor-loading operations were at various times under the supervision of T. W. Fehr, A. S. Ginsburgh, or L. B. Loehr. Static testing and data reduction were supervised by R. P. Meeker and R. R. Stephens, respectively. Additional ballistic data were supplied by R. E. Davis and W. R. Kirchner.

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(*Continued on page 26*)

Charge Geometry and Ballistic Parameters for Solid Propellant Rocket Motors

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The dependence of four important parameters of internal ballistics of solid propellant rockets on charge and motor geometry (during burning) is derived for the case of straight, side burning charges with linear variation of surface area relative to distance burned in. The parameters considered are the ratio of burning surface area to nozzle throat area, burning surface area to internal channel area, nozzle throat area to internal channel area, and charge mass. The analysis uses dimensionless variables in order to generalize results and exhibit scale effects. Application of the results to an elementary theory of motor performance is demonstrated.

Nomenclature

a = fractional change in surface area, Equation [2]
A_e = cross-sectional area of the propellant charge
A_m = cross-sectional area of the interior of the motor tube
A_p = cross-sectional area of the flow channel adjacent to the charge, $A_m - A_e$
A_t = cross-sectional area of the exhaust nozzle at its throat
b = $q/\pi D$
C = factor in a burning rate rule, usually dependent on propellant temperature
C_D = constant in the discharge equation for a sonic nozzle
C_F = factor in the thrust equation for a rocket motor, dependent on nozzle expansion ratio and the ratio of motor pressure to atmospheric pressure
d = web thickness of charge (distance remaining to be burned)
D = inside diameter of the motor tube
F = thrust of a rocket motor
G = internal area ratio S_c/A_p
I = impulse of a rocket motor
I_{sp} = specific impulse, I/M_i
J = channel area ratio, A_t/A_p
K = nozzle area ratio, $S_c/A_t = G/J$
l = L/D
L = length of the propellant charge
m_d = mass discharge rate through a sonic nozzle, $C_D p A_t$
M = mass of solid propellant in a motor
n = pressure exponent in the burning rate rule $r = Cp^n$
p = pressure in the combustion chamber (assumed uniform)
P = $\lambda_1 p$
q = perimeter of the cross section of the burning surface
r = burning rate of a solid propellant (linear)
r_i = ratio of initial inside radius to outside diameter of a circular cylindrical propellant charge
R = loading density of a motor, A_e/A_m
s = distance burned into a propellant charge from the original surface
S_c = burning surface area of a charge
t = time from the start of burning of a charge
T = $\lambda_2 t$
V = volume of a propellant charge
w = d/D
W = $w/w_i = d/d_i$
λ_1 = scale factor relating P and p (see Equation [16])
λ_2 = scale factor relating T and t (see Equation [17])
ρ_p = density of the solid propellant

Subscripts

f = conditions at the end of burning
i = conditions at the start of burning

1 Introduction

A DISTINCTIVE feature of solid propellant rocket motors is the control of mass flux and pressure, and hence of thrust, by the geometrical properties of the propellant charge. Thus the thrust programming is done by the designer of the rocket and is not subject to subsequent modification except by the "unsolicited" effects of temperature and the rarely exploited effect of controlled variation of nozzle size during operation. For the majority of current ordnance rocket applications this characteristic of solid propellant rocket technology is not a disadvantage. However, it is desirable for the engineer to be able to design a propellant charge so that it will produce a specified thrust program, without too much trial-and-error development, and it is desirable for the workers in other phases of rocket technology to know what kind of thrust-time programs are feasible and what relations exist among the forms of the thrust-time program, the mass-time program, and the configuration of the rocket.

To achieve these objectives, it is necessary to have a theory of motor performance which is, on the one hand, sufficiently general to indicate the important aspects of behavior with some realism, and, on the other hand, sufficiently elementary to make it easily accessible to the persons needing it.

The objective of the present paper is to examine the dependence of conventional parameters of internal ballistics of solid propellant rockets on the geometry of the propellant charge. In this case the restrictions made represent a compromise between reality and simplicity aimed at rough quantitative prediction of motor performance under unconventional conditions and rather accurate prediction under conventional conditions. When coupled with the results of an earlier analysis of the gas flow field in a motor in terms of these ballistic parameters (1),[2] the present paper will permit a relatively simple evaluation of motor performance in terms of propellant charge design.

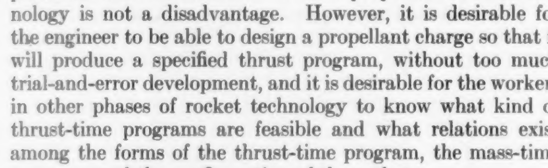

FIG. 1 ARRANGEMENT OF PARTS OF A SOLID PROPELLANT MOTOR

The essential features of a solid propellant rocket motor are shown in Fig. 1, and some of the notation of this paper is identified there. The motor consists simply of a suitable high-pressure container for the propellant charge, with a Laval nozzle to accelerate the escaping gases to the highest velocity consistent with the pressure of the surrounding atmosphere. The charge is so designed that all exposed surfaces are connected by suitable flow channels to the nozzle, to pro-

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[2] Numbers in parentheses refer to References on page 21.

vide for escape of combustion gases. Burning is found to proceed inward from the initial surface of the propellant charge at a predictable rate, roughly the same rate everywhere that propellant surface is exposed. In order to obtain the desired programming of gas evolution, charges are designed with many different shapes. Some surfaces may be coated with noninflammable material to prevent burning, thus providing an additional design variable for programming the burning. This "inhibiting" also provides nonburning surfaces for support of the charge during operation. A number of different charge geometries are described by Wimpress (2, chapter 8). It is not intended to discuss charge shapes in detail here, but rather to approximate all charges by certain programs of burning surface variation during operation.

The most important assumptions in the present paper are the following: (a) The propellant charge is cylindrical in the general sense (i.e., it is straight and its cross-sectional shape is the same over its entire length) and remains so during burning, with no burning on the ends; (b) the area of the burning surface is a linear function of the distance burned into the web from the initial surface.

These assumptions are fairly well satisfied by most motors now in use, except those using end-burning propellant charges. What may be more important, from the standpoint of utility of the results, is that there seems to be very little prospect of a broad analysis of the relation of motor performance to ballistic parameters without assumption (a), while approximation (b) is more general than that used in most analyses of this type and yet simple enough to result in tractable relations between charge geometry and ballistic constants. The charge shape is also determined in some measure by available methods of fabrication, and assumption (a) is consistent with methods currently in use (e.g., extrusion and casting).

Assumptions (a) and (b) enable one to define the geometrical characteristics of a propellant charge at any instant during burning by the length L, which is assumed constant; the cross-sectional area A_e, which is assumed the same over the entire length of the charge; the thickness of the web d, which is the distance remaining to be burned in before burning is completed, and is assumed to be the same everywhere on the charge; uninhibited portion of the perimeter of a cross section of the charge q, which is assumed to be the same everywhere along the charge; the burning surface area S_c, assumed to vary linearly with d during burning; the volume, V, of solid propellant. The assumptions noted in the foregoing are all consequences of assumptions (a) and (b). The pertinent geometrical properties which can be inferred from the above variables and assumptions can all be found in the properties of a trapezoidal bar of length L and cross section as depicted in Fig. 2 (the heavy outlines represent inhibited surfaces). From this figure the following simple relations may be inferred between the variables defined above,

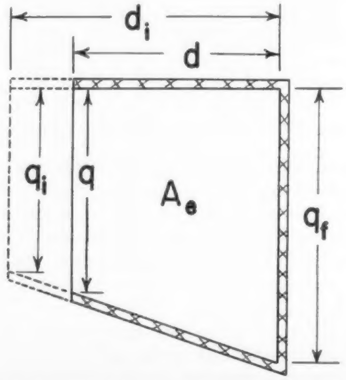

FIG. 2 GEOMETRICAL MODEL FOR A PROPELLANT CHARGE

where the subscript i indicates conditions at the start of burning of the charge and the subscript f indicates conditions at the finish of burning of the charge.

$$q = q_i[1 - a(d_i - d)/d_i] \qquad [1]$$

$$a \equiv \frac{q_i - q_f}{q_i} \qquad [2]$$

$$S_c = qL \qquad [3]$$

$$A_e = d(q + q_f)/2 \qquad [4]$$

$$V = A_e L \qquad [5]$$

In rather specific terms, the object of this paper will be to determine the relationships among the foregoing dimensional variables and four important ballistic parameters, as noted in the titles of the following sections. In the analysis, the loading density $R = A_e/A_m$ will be used extensively, where A_m is the cross-sectional area of the inside of the motor tube (assumed the same over the length of the charge), and A_e is the cross-sectional area of the propellant charge. Under the assumptions of (1), knowledge of two area ratios G and J and of certain nongeometrical properties of the propellant is sufficient to predict the pressure in the rocket. Further choice of nozzle geometry establishes the thrust of the motor, while knowledge of the mass M of the propellant and the mass of inert parts of the motor then is sufficient to predict acceleration of the rocket, provided usual considerations of drag and gravity forces are made.

In the next section, the geometrical variables described above will be changed to dimensionless form, and in subsequent derivations they will be lumped where possible, in order to classify more clearly the role of each variable. Using the dimensionless variables, there follows a sequence of derivations of ballistic parameters in terms of geometrical variables, derivations from which most of the tedious algebraic manipulation has been deleted. The particular choice of independent variables made in these analyses was made on the basis of their relatively high degree of restriction by motor performance requirements or by design and fabrication limitations. For certain design problems the reader may find Equations [6] and [7] useful for introduction of different variables.

2 Ballistic Parameters in Terms of Charge and Motor Shape

Dimensionless Variables

In order to generalize the analysis as much as possible, the variables describing the charge geometry will ordinarily be made nondimensional by division by a characteristic dimension of the motor. Then it will always be possible to interpret the results in terms of a certain motor simply by inserting the appropriate dimensional factor in the geometrical variable. Simple functions of the outside diameter of the motor tube are used as the characteristic dimensions. The charge geometry is then expressible in terms of the following variables

D = outside diameter of charge

$b = q/\pi D$ = burning perimeter of cross-section shape

$l = L/D$ = length of charge

$R = A_e/A_m$ = loading density

$w = d/D$ = web thickness

The changes of these variables during burning of the charge are of considerable interest and will be used in the following sections. In particular, the following relations will be used repeatedly.

$$w_i = \frac{R_i(1-a)}{2b_f(2-a)} \qquad [6]$$

$$b_i = b_f/(1-a) \qquad [7]$$

$$b = \frac{b_f}{R_i(1-a)^2}\,[R_i(1-a)^2 + 2b_f a(2-a)w] \qquad [8]$$

$$R = \frac{4b_f w}{R_i(1-a)^2}\,[R_i(1-a)^2 + b_f a(2-a)w] \qquad [9]$$

Ratio of Burning Surface Area to Nozzle Throat Area

A ballistic parameter which is used extensively in rocket work and is particularly applicable when high gas velocities do not prevail in the motor interior except in the nozzle, is the ratio K of the charge area which is liberating mass to the nozzle throat area which is discharging mass. Here the nozzle throat area A_t will be expressed only in the combination A_t/A_m in order to generalize to all sizes of motors. Under the assumptions of the present report

$$K \equiv S_c/A_t = \frac{S_c}{A_m}\Big/\frac{A_t}{A_m} = \frac{qL}{A_m}\Big/\frac{A_t}{A_m} = \frac{\pi D^2 bl}{A_m}\Big/\frac{A_t}{A_m} \qquad [10]$$

From the relations [8] and [10], K can be expressed as follows

$$K = 4lb_f[R_i(1-a)^2 + 2a(2-a)b_f w]/[R_i(1-a)^2][A_t/A_m] \qquad [11]$$

The choice of variables in deriving Equation [11] is, of course, somewhat arbitrary and is justified largely by its utility in terms of rocket problems in general. Thus the variable R_i is the loading density of the motor cross section and is important in determining performance characteristics dependent on mass ratio of the rocket such as velocity and range. The variable a is indicative of thrust programming. The factor l is the ratio of length to diameter of the charge, and is often important in determining the stability of the missile in flight. The quantity b_f is the final perimeter of the burning surface, a geometrical variable which is usually subjected by practical considerations to a relatively small range of values near 1, and is intimately associated with reaction time within that range of values. The variable w is the thickness of the charge web (one half the web thickness when burning takes place on both sides of the web), and its value during burning is descriptive of the time of burning.

The form of Equation [1] is considerably simplified if the variable w is transformed by the following relation

$$w = \frac{w}{w_i}\,w_i = W\,\frac{R_i(1-a)}{2b_f(2-a)} \qquad [12]$$

where the variable W is the fractional web thickness at any time during burning. Then Equation [11] becomes

$$K = 4lb_f[1 + a(W-1)]/[1-a][A_t/A_m]$$

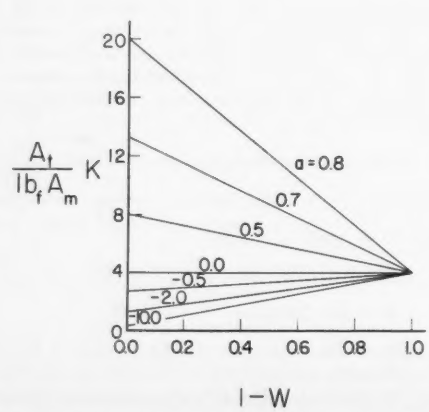

FIG. 3 DEPENDENCE OF NOZZLE AREA RATIO ON CHARGE GEOMETRY

If this relation is written as in [11a], it can be summarized graphically, as in Fig. 3, as a family of straight lines.

$$\frac{K}{lb_f}\,\frac{A_t}{A_m} = 4\,\frac{1 + a(W-1)}{1-a} \qquad [11a]$$

Application of Equation [11a] to determination of pressure-time curves in motors with linear surface charges is demonstrated in section 3. It is significant that the quantities A_t, l, b_f, and A_m enter into Equation [11a] only in the factor A_t/lb_fA_m, and that the quantity R_i does not enter into the relation at all. The value of K at the end of burning (and throughout burning when $a = 0$) is

$$K_f = 4lb_f A_m/A_t \qquad [11b]$$

Ratio of Burning Surface Area to Internal Channel Area

A second ballistic parameter (one which assumed an important role in determining motor performance in Ref. 1), is the ratio

$$G = S_c/A_p$$

For the assumptions of the present paper, it can be shown that

$$G = \frac{4lb_f\, 2ab_f(2-a)w + R_i(1-a)^2}{4ab_f^2(a-2)w^2 + R_i(1-a)^2(1-4b_f w)} \qquad [13]$$

and for the special cases of the finish and start of the burning period (respectively)

$$(G/l)_f = 4b_f \qquad [13a]$$

$$(G/l)_i = \frac{4b_f}{(1-a)(1-R_i)} \qquad [13b]$$

Also, when $a = 0$

$$(G/l)_{a=0} = \frac{4b_f}{1 - 4b_f w} \qquad [13c]$$

Introducing the variable W in Equation [13], as in Equation [11], the following relation results, indicating the variation in G during burning

$$G/lb_f = \frac{4(a-2)}{1-a}\;\frac{1 + a(W-1)}{aR_iW^2 + 2(1-a)R_iW + a - 2} \qquad [13d]$$

The number of variables (W, a, R_i) on the right in Equation [13d] is such that the relation can be summarized by a series of graphs with R_i constant in each graph, where a graph consists of several curves of G/lb_f vs. W with various values of a. Such a series of curves is presented in Fig. 4. It is significant that the dimensions l and b_f enter into Equation [13d] only as the factor lb_f. This means that, on the time scale implied by the variable W, G depends upon l and b_f only through the factor lb_f, and l and b_f both enter into the relation in the same way. This statement is somewhat deceptive because the fractional scale implied by W hides the changes in w_i which are taking place with changes in R_i, a, and b_f according to Equation [6].

Ratio of Nozzle Throat Area to Internal Channel Area

A third ballistic parameter, denoted in (1) by the symbol J, is the ratio of the nozzle throat area A_t to the area A_p of the flow channel in the motor. In terms of one-dimensional, isentropic channel flow, this ratio is indicative of the pressure, density, and velocity ratios of the stream at the downstream end of the propellant charge relative to conditions in the channel at the nozzle throat. When the ratio is large, i.e., near 1, the velocities of the gas in the motor are large, the pressure differences are large, the equilibrium conditions in the motor are relatively sensitive to A_p, the thrust program is very different from the pressure program, and the ballistic

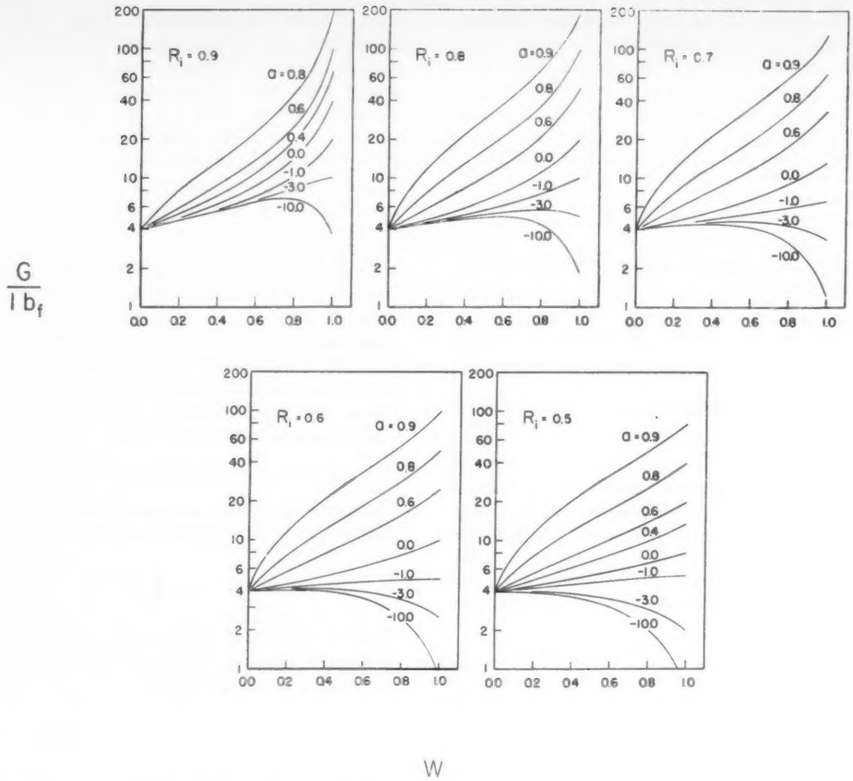

FIG. 4 DEPENDENCE OF INTERNAL AREA RATIO ON CHARGE GEOMETRY

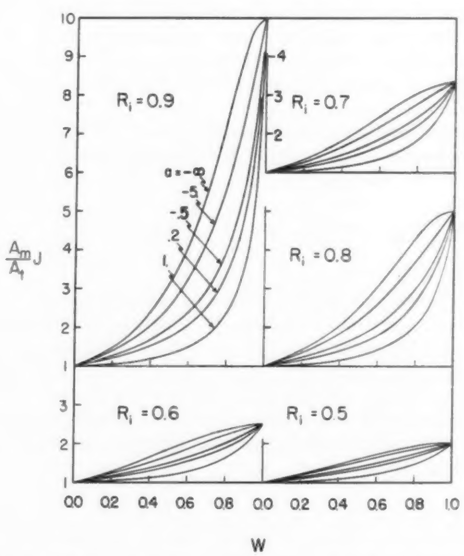

FIG. 5 DEPENDENCE OF CHANNEL AREA RATIO ON CHARGE GEOMETRY

parameter K is not sufficiently descriptive of motor-charge geometry to establish the flow field. However, it was shown in (1) that even with large values of J the flow field is largely determined by the two geometrical parameters G and J.

To compute J as a function of the variables used for K and G, write

$$J = A_t/A_p = \frac{A_t}{A_m}\Big/\frac{A_p}{A_m} = \frac{A_t}{A_m}\Big/[1 - R]$$

where A_t has been divided by A_m to generalize relative to motor size. When Equation [9] is combined with the preceding, one may obtain

$$\frac{A_m}{A_t}J = \frac{1}{1 - R} = \frac{R_i(1 - a)^2}{R_i(1 - a)^2(1 - 4b_f w) - 4b_f^2 a(2 - a)w^2} \quad \text{[14]}$$

In this expression, when $w = 0$

$$(A_m/A_t)J = 1 \quad \text{[14a]}$$

when $w = w_i$

$$(A_m/A_t)J = 1/(1 - R_i) \quad \text{[14b]}$$

when $a = 0$

$$(A_m/A_t)J = 1/(1 - 4b_f w) \quad \text{[14c]}$$

or, introducing the variable W in place of w

$$\frac{A_m}{A_t}J = \frac{2 - a}{2 - a + R_i[aW(2 - W) - 2W]} = \frac{(1 - a)}{4[1 + a(W - 1)]}\frac{G}{lb_f} \quad \text{[14d]}$$

In Fig. 5, the dependence of $(A_m/A_t)J$ on W is shown for several combinations of a and R_i.

Charge Mass

In some rocket motors ($J < 0.4$) the variation of K during burning may provide sufficient information regarding charge geometry to determine operating pressures. In most cases, knowledge of two parameters such as G and J is sufficient. In all cases, prediction of motor performance requires, in addition: (a) the nozzle geometry and atmospheric pressure in order to determine the thrust, and (b) the rocket mass (including unburned propellant) to determine acceleration.

Item (b) is a matter, in part, of charge geometry, since the

mass of the unburned propellant at any time during burning can be computed from the charge geometry.

With the geometrical assumptions of the present paper, it is possible to express the charge mass as a relatively simple function of the variables used in the last sections. Suppose the density of the solid propellant is ρ_p and the charge mass and volume are denoted by M and V, respectively. Then

$$M = \rho_p V = \rho_p A_c L = \rho_p \frac{q + q_f}{2} dL$$

Changing to dimensionless variables and substituting Equation [8] and then Equation [9]

$$M = \pi D^3 \rho_p \frac{lwb_f}{R_i(1 - a)^2} [R_i(1 - a)^2 + b_f a(2 - a)w] = \pi D^3 \rho_p Rl/4 \ldots\ldots\ldots\ldots [15]$$

Changing to the variable W, this becomes

$$M = \pi D^3 \rho_p l R_i [a(W - 2) + 2]W/(2 - a) \ldots\ldots [15a]$$

For some purposes it is convenient to make this result dimensionless by dividing both sides of the equation by the mass of propellant the motor would hold if it were 100 per cent full of propellant in the section along the length of the charge. This dimensionless quantity is the loading density R for the present geometry, and is given in terms of the fractional web thickness W by

$$R = R_i [a(W - 2) + 2]W/(2 - a) \ldots\ldots\ldots [15b]$$

Variation in the quantity R/R_i with W is shown in Fig. 6 for

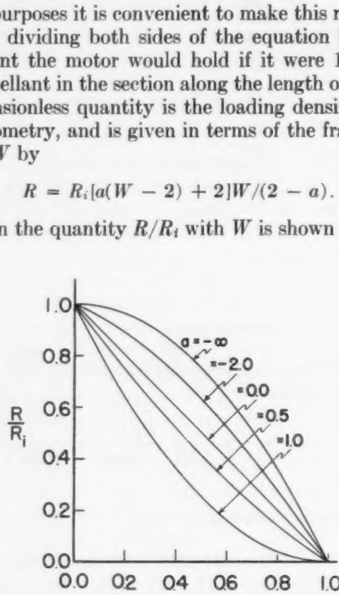

FIG. 6 DEPENDENCE OF CHARGE MASS ON CHARGE GEOMETRY

various values of a. This is the ratio M/M_i of charge mass to initial charge mass.

3 Interpretation and Application of Results

General Discussion

The analysis in this paper is similar to the determination of form functions in internal ballistics of guns (3). The ballistic parameters in the present case are the dependent variables K, J, G, and R. The variation of these dependent variables during burning of the solid propellant charge determines the way in which the pressure (and thrust) of the rocket motor varies during burning (1). The quantities a, R_i, b_f, l, and A_t/A_m have been shown to be sufficient to establish the variation of these ballistic parameters for the case of straight-sided charges having linear variation of burning area with distance burned in. Since most charge configurations can be approximated by such a geometry, the present results will prove useful in qualitative evaluations of rocket potentialities and in preliminary selection of internal design. The dimensionless nature of variables indicates to what extent performance of motors is independent of size. An obvious next step in refinement of the analysis is to assume a more general form of dependence of burning area on distance burned into the charge web, such as a parabolic relation. It may also be advantageous to work out the dependence of the ballistic parameters on W for specific charge configurations such as those described by Wimpress (2, cf. 8).

Application of Results in Elementary Internal Ballistic Theory

As an illustration of application of the present results, the performance of a motor with linear surface charge was computed using an internal ballistic theory somewhat simpler than that of Ref. 1. Thus, if the variations in conditions in different parts of the rocket combustion chamber are negligible, the equilibrium between mass burning rate of the charge and mass discharge rate of the nozzle is described by the familiar equation (2, p. 76)

$$\rho_p S_c r = C_D A_t p$$

From this equation, assuming that the propellant burning rate is given by the typical relation $r = Cp^n$ (2, p. 17)

$$p = \left(\frac{\rho_p C}{C_D} K\right)^{1/(1-n)}$$

Then if the charge geometry is subject to the restrictions of the present report, Equation [11a] may be combined with the last two equations to give

$$P = \lambda_1 p = \left(\frac{4lb_f\rho_p}{C_D}\frac{CA_m}{A_t}\right)^{-1/(1-n)} p = \left(\frac{1 + a(W - 1)}{1 - a}\right)^{1/(1-n)} \ldots [16]$$

where the quantities P and λ_1 are defined by the equation, and P is proportional to the operating pressure during burning. The proportionality factor contains all of the design and propellant variables except a and n, so that the fractional variation in pressure during burning depends only on a, n, and W.

The time during burning can be related to W by integrating the relation $dt = ds/r$, where ds is the differential of distance burned into the charge web. The differential ds can be written

$$ds = -d(Dw) = -d(Dw_i W) = -Dw_i dW = -[R_i(1 - a)D/2b_f(2 - a)]dW$$

If this is combined with the burning rate rule $r = Cp^n$, and p is substituted from Equation [16], dt can be evaluated in terms of the same variables as used in previous equations. The resulting expression for dt was integrated with respect to W between the limits 1 and W, and the relation obtained was used to define a quantity T which is proportional to the time of burning t, and a function only of a, n, and W. The resulting expression for T was [3]

$$T = \lambda_2 t = \frac{1}{DR_i}\frac{1 - 2n}{n - 1}\left[\frac{1}{Cb_f 2^{n+1}}\left(\frac{l\rho_p A_m}{C_D A_t}\right)^{-n}\right]^{-1/(1-n)} t$$

$$= \left[\frac{1 - (1 + a[W - 1])^{(1-2n)/(1-n)}}{a(2 - a)(1 - a)^{-1/(1-n)}}\right] \ldots\ldots\ldots [17]$$

The dependence of P on T through the parameter W was com-

[3] The function for t is indeterminate if $n = 1/2$ or $a = 0$. However, application of L. Hospital's rule to the case where $n = 1/2$ gives for the indeterminate factor

$$\lim_{n \to 1/2} \frac{1 - (1 + a[W - 1])^{(1-2n)/(1-n)}}{1 - 2n} = 2ln(1 + a[W - 1])$$

For the case where $a = 0$

$$\lim_{a \to 0} T = \frac{(2n - 1)(w - 1)}{2(1 - n)}$$

puted for several values of a with $n = 0.55$. These results are shown in Fig. 7. Under the restrictions made up to this point, the curves in Fig. 7 are the possible pressure-time curves for solid propellant rocket motors, except that the P and T scales must be shrunk or expanded according to the scale factors λ_1 and λ_2 indicated in Equations [16] and [17].

The fact that the curves in Fig. 7 all terminate at the same value of P is because b_f was used as one of the design variables.

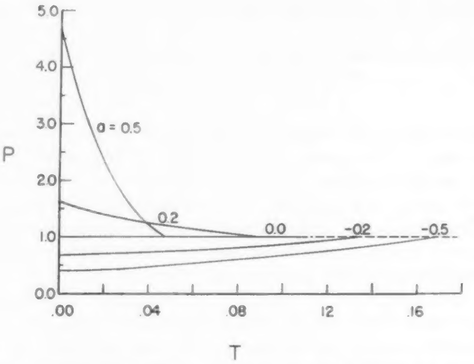

FIG. 7 PRESSURE-TIME RELATIONSHIPS FOR LOW-J MOTORS

The area under a curve of Fig. 7 is proportional to the pressure-time integral with the proportionality factor indicated by

$$\int P\,dT = \lambda_1\lambda_2 \int p\,dt$$

If the value of $\int_{W=1}^{W=0} P\,dT$ is computed from Equations [16] and [17] by direct integration, the result

$$\int_{W=1}^{W=0} P\,dT = \frac{\lambda_1{}^n\lambda_2 R_i D}{4b_f C}$$

is obtained. The absence of a in this expression implies that the areas under the curves in Fig. 7 are all the same, and that the pressure-time integral depends on one less geometrical variable than do the "pressure" P and the "reaction time" T (in this treatment, the geometrical variable a was the one eliminated). From this result one can write the pressure-time integral as

$$\int p\,dt = \frac{\lambda_1{}^{n-1}R_i D}{4b_f C} = \frac{\rho_p}{C_D}\frac{A_m}{A_t} LR_i = \frac{M_i}{C_D A_t} \quad \ldots\ldots [18]$$

The last result on the right could have been inferred more directly by integrating the nozzle discharge rate relation $m_d = C_D A_t p$ over the reaction time of the motor; but the procedure used above has some appeal as a logical extension of the results in Equations [16] and [17].

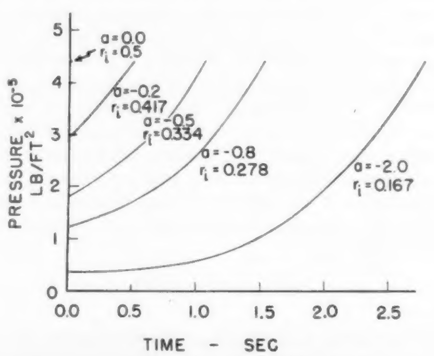

FIG. 8 PRESSURE-TIME CURVES FOR INTERNAL-BURNING, CIRCULAR CYLINDRICAL CHARGES

From Equation [18] and the thrust relation $F = C_F A_t p$ (2, p. 43) it follows that the impulse of the motor is

$$I \equiv \int F\,dt = \frac{C_F}{C_D} M_i = I_{sp}M_i = I_{sp}\rho_p L A_m R_i$$

where the variation in C_F due to pressure changes during burning has been neglected. This equation illustrates the importance of the geometrical variable R_i, and emphasizes the independence of I of the charge area variation designated by the variable a. The choice of a value for a usually depends on external considerations such as desired or optimum acceleration program, optimum launching conditions, or on motor design requirements such as a regressive pressure-time function to match the decline in motor component strength resulting from progressive temperature rise during operation.

Application of Results Illustrated

As a particular elementary example of applications indicated in the previous section, consider the behavior of motors with charges which are right circular cylindrical tubes burning only on the internal wall with external diameter equal to D. For such a geometry the following conditions hold

$$a = \frac{2r_i - 1}{2r_i} \quad \ldots\ldots\ldots\ldots\ldots\ldots [19]$$

$$R_i = 1 - 4r_i{}^2 \quad \ldots\ldots\ldots\ldots\ldots\ldots [20]$$

where r_i is the initial inside radius of the charge divided by D. Now λ_1 and λ_2 of Equations [16] and [17] are dependent on additional details of geometry and propellant, but P and T are dependent only on r_i, n, and W as indicated by [16], [17], and [19]. Thus [19] determines which curve in Fig. 7 is pertinent in the present case, and hence what the shape of the pressure-time curve is. When sufficient information is specified to establish λ_1 and λ_2, Fig. 7 may be converted to true pressure-time coordinates.

Typical values of the factors in λ_1, and λ_2 are

$l = 10.0$

$b_f = 1.0$

$\rho_p = 3.0$ slug/ft^3

$C = 6.1 \times 10^{-5}$ ft^{2n+1} lb^n sec^{-1}

$C_D = 2.11 \times 10^{-4}$ slug/lb sec

$A_m/A_t = 10.0$

$n = 0.55$

$D = 0.20$ ft

Using these numbers, λ_1 and λ_2 were evaluated so that Equations [16] and [17] could be written

$$p = P/\lambda_1 = 4.41 \times 10^5 P$$

$$t = T/\lambda_2 = 5.78(1 - 4r_i{}^2)T$$

Using the foregoing with Fig. 7 and Equation [19], the pressure-time curves for various values of r_i were computed and are shown in Fig. 8 to illustrate application of the P-T functions of Fig. 7.

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The Correlation of Interior Ballistic Data for Solid Propellants

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A self-consistent set of empirical equations is presented for describing concisely the most important performance parameters of a solid propellant rocket motor. These equations relate the propellant burning rate, propellant temperature, chamber pressure, and area ratio so that the designer can construct a coherent set of correlations from only fragmentary data. The problem of propellant variability also is examined, and a number of relations among the variances of the various parameters are obtained.

Introduction

THE interior ballistic properties of solid propellants for rocket motors are usually determined from static firings of a series of propellant specimens at a constant initial temperature, T, in a standard motor with a different nozzle-throat diameter for each specimen. For each firing one obtains the chamber pressure, p (1000 lbf in.$^{-2}$), the linear burning rate of the propellant, r (in sec^{-1}), the mass flow coefficient, $C_w = w/pA_t$ (lbm lbf^{-1} sec^{-1}), and the area ratio $K = A_p/A_t$. In these definitions w is the mass flow rate through the nozzle (lbm sec^{-1}), A_t is the nozzle-throat area (in.^{2}), and A_p is the propellant burning area (in.^{2}). Similar sets of data for different propellant temperatures are derived from other series of propellant specimens. The data so obtained permit the determination of r, C_w, and K, each as a function of p and T, in a form suitable for design purposes.

Less inclusive data are obtainable from rocket manufacturing records. In the manufacture of a large number of rockets of one design, it is the usual practice to fire statically a specified proportion of the rockets produced. Such firings are generally made at several propellant temperatures but with only one nozzle-throat diameter, i.e., at constant K. The reproducibility of r and p is then the principal item of interest.

Correlation and interpretation of either type of data can be confusing because of the rather intricate manner in which r, C_w, K, and p are interrelated. However, these relationships can be turned to advantage in providing a means of checking and proving the internal consistency of the data. Furthermore, such relationships make it possible to derive a large amount of information from a small amount of experimental work. The purpose of the present discussion is to set forth a convenient system for the correlation of both types of data.

Temperature Coefficients

In the absence of erosive burning it is usually assumed that the propellant burning rate is a power function of the pressure of the gas in contact with the burning surface, i.e.,

$$r = cp^n \qquad [1]$$

Here, c and n are constants independent of p, and n is less than unity. The general applicability of Equation [1] has been amply verified for many different propellants. In special cases, the equation is not applicable, but these are rare if attention is confined to a moderate range of pressure. Different values of the constants may then be used for individual portions of the log r vs. log p curve.

It is convenient to assume that the effect of propellant temperature T on burning rate is occasioned by changes in the parameter c and not by changes in the parameter n. Although this assumption has not been adequately investigated because of the experimental difficulties in establishing n compared to c, it has at least described the known facts within experimental error. A number of different mathematical formulas have been used to represent the dependence of c upon T, but the following one is especially desirable because of the simplicity of the relations derivable from it.

$$c = c_0e^{u(T - T_0)} \qquad [2]$$

Here, c_0 and u are empirical constants independent of T. The constant c_0 is simply the value of c at the standard propellant temperature T_0, and the constant u is the relative temperature coefficient of c derived by partially differentiating Equation [2]

$$\frac{1}{c}\left(\frac{\partial c}{\partial T}\right) = u \qquad [3]$$

Inasmuch as burning rates increase with increasing propellant temperature, u will be positive.[3] It should be understood that there is no a priori reason why u should be constant, but the usefulness of Equation [2] will be limited unless this is found to be the case, at least over specific ranges of propellant temperature.

Another equation empirically chosen for its usefulness relates the mass flow coefficient, defined by $w = C_wA_tp$, to the chamber pressure

$$C_w = hp^q \qquad [4]$$

Here h and q are constants independent of p, and because C_w decreases with increased chamber pressure, q will be negative. The applicability of Equation [4] has not been as extensively verified as Equation [1] and it has not found widespread usage. Nevertheless, in some instances there is a definite gain in accuracy to be achieved by taking into account the variability of C_w with p, and Equation [4] is adequate for the purpose.

It is again assumed that the exponent is a constant under all circumstances and the effect of temperature is limited to changes in the parameter h, which may be represented by an equation of the form

$$h = h_0e^{v(T - T_0)} \qquad [5]$$

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[3] It may be noted that u is the temperature coefficient of burning rate at constant pressure $(\partial \ln M/\partial T)_p$ for which theoretical expressions are given in the literature. See Refs. 1 and 2 listed at end of the paper.

Here, h_0 and v are empirical constants independent of T. The constant h_0 is simply the value of h at the standard propellant temperature T_0, and the constant v is the relative temperature coefficient of h derived by partially differentiating Equation [5]

$$\frac{1}{h}\left(\frac{\partial h}{\partial T}\right) = v \qquad [6]$$

It should be understood that there is likewise no a priori reason why v should be constant, but the usefulness of Equation [5] will be limited unless this is found to be the case at least over specific ranges of propellant temperature. Actually, v is practically indistinguishable from zero in the cases so far investigated.

A mass balance around a solid propellant rocket motor under steady-state conditions gives

$$rA_p\rho = C_w A_t p$$

where A_p is the propellant burning area (in.^{2}) and ρ is the density of the solid propellant minus the density of the propellant gas (lbm in.$^{-3}$). This equation may be written in the form

$$K = C_w p / r\rho$$

and substitution of Equations [1] and [4] gives

$$K = (h/c\rho)p^{1-n+q}$$

If this dependence of K on p is described empirically by the equation

$$K = gp^m \qquad [7]$$

as is customary, then it is apparent that the following relations must hold

$$g = h/c\rho \qquad [8]$$

$$m = 1 - n + q \qquad [9]$$

Furthermore, if the temperature dependence of K on p is ascribed solely to variations in g according to a representation of the form

$$g = g_0 e^{w(T - T_0)} \qquad [10]$$

the one obtains

$$g = \frac{h_0}{c_0\rho} e^{(v - u)(T - T_0)} \qquad [11]$$

and the relative temperature coefficient of g is

$$\frac{1}{g}\left(\frac{\partial g}{\partial T}\right) = w = v - u \qquad [12]$$

Obviously w will be negative.

The effect of propellant temperature upon rocket motor operation is easily deducible from the foregoing equations. Equation [7] can be written

$$p = (K/g)^{1/m} \qquad [13]$$

and differentiation with respect to propellant temperature at constant K gives

$$\pi_p \equiv \frac{1}{p}\left(\frac{\partial p}{\partial T}\right)_K = -\frac{1}{m}\left(\frac{\partial g}{\partial T}\right)\frac{1}{g} \qquad [14]$$

$$= -w/m \qquad [15]$$

$$= (u - v)/m \qquad [16]$$

Differentiation of Equation [4] gives the equations

$$\frac{1}{C_w}\left(\frac{\partial C_w}{\partial T}\right)_p = \frac{1}{h}\left(\frac{\partial h}{\partial T}\right) \qquad [17]$$

$$= v \qquad [18]$$

$$\pi_C \equiv \frac{1}{C_w}\left(\frac{\partial C_w}{\partial T}\right)_K = \frac{1}{h}\left(\frac{\partial h}{\partial T}\right) + \frac{q}{p}\left(\frac{\partial p}{\partial T}\right)_K \qquad [19]$$

$$= v - (q/m)w \qquad [20]$$

$$= (q/m)u + (1 - q/m)v \qquad [21]$$

Differentiation of Equation [1] gives the equations

$$\frac{1}{r}\left(\frac{\partial r}{\partial T}\right)_p = \frac{1}{c}\left(\frac{\partial c}{\partial T}\right) \qquad [22]$$

$$= u \qquad [23]$$

$$\pi_r \equiv \frac{1}{r}\left(\frac{\partial r}{\partial T}\right)_K = \frac{1}{c}\left(\frac{\partial c}{\partial T}\right) + \frac{n}{p}\left(\frac{\partial p}{\partial T}\right)_K \qquad [24]$$

$$= u - (n/m)w \qquad [25]$$

$$= (1 + n/m)u - (n/m)v \qquad [26]$$

It will be noted that the two modes of differentiation, with K constant and with p constant, correspond to two methods of experimentation—the former to motor study with firings at a constant nozzle size, and the latter to propellant study with firings at varying nozzle size. The three relative temperature coefficients π_p, π_C, and π_r, derived from firings of a given motor at various temperatures, are obtained by plotting p, C_w, and r vs. temperature on semilog paper. The slopes are related by the equation

$$\pi_p = \pi_r - \pi_C \qquad [27]$$

In rocket design, the temperature coefficients needed are π_p, to obtain the chamber pressures at the upper and lower limits of firing temperature, and π_r, to obtain the duration of burning at the upper and lower limits of firing temperature.

The parameters u, v, and w may be termed "inherent" relative temperature coefficients inasmuch as they are characteristic of the propellant in question without reference to the effect of operation in a motor as expressed by the mass balance equation. They may be obtained from π_p, π_C, and π_r by means of Equations [15], [21], and [26] (although the reverse procedure will be utilized more often), or alternatively by cross-plotting on semilog paper the usual graphs of log K vs. log p, log C_w vs. log p, and log r vs. log p at several temperatures and at constant pressure. A summary of the foregoing relations is given in Table 1.

TABLE 1 SUMMARY OF TEMPERATURE COEFFICIENTS

Lines at constant T log-log paper	Lines at constant K semilog paper	Lines at constant p semilog paper	Check ↓
log K vs. log p $K = gp^m$	log p vs. T	log K vs. T $g = g_0e^{w(T - T_0)}$	
slope $= m$	slope $= \pi_p$	slope $= w$	$w = -m\pi_p$
log C_w vs. log p $C_w = hp^q$	log C_w vs. T	log C_w vs. T $h = h_0e^{v(T - T_0)}$	
slope $= q$	slope $= \pi_C$	slope $= v$	$v = \pi_C - q\pi_p$
log r vs. log p $r = cp^n$	log r vs. T	log r vs. T $c = c_0e^{u(T - T_0)}$	
slope $= n$	slope $= \pi_r$	slope $= u$	$u = \pi_r - n\pi_p$
$g = h/c\rho$	$\pi_p = \pi_r - \pi_C$	$g_0 = h_0/c_0\rho$	← Check
$m + n - q = 1$		$w = v - u$	

Performance Variability

The derivations up to this point have assumed a unique, invariable set of quantities characterizing a given propellant

at constant temperature, namely, c, g, h, n, m, and q. However, in practice it is found that a certain amount of variability in these quantities from one sample of propellant to another must be taken into account. Inasmuch as reproducibility in propellant performance is of utmost importance in manufacturing rockets economically, a measure of variability is necessary. The variability is most conveniently expressed in terms of the estimated standard deviations of the ballistic parameters, and the relations between the various estimated standard deviations can be found by utilizing the law of propagation of error for uncorrelated variables. This law states that if

$$f = f(x, y, z)$$

then

$$\sigma_f^2 = \left(\frac{\partial f}{\partial x}\right)^2 \sigma_x^2 + \left(\frac{\partial f}{\partial y}\right)^2 \sigma_y^2 + \left(\frac{\partial f}{\partial z}\right)^2 \sigma_z^2$$

In the particular case where

$$f = xy^a$$

and when the standard deviations are expressed in fractional or percentage values, symbolized by $s_x = 100\sigma_x/x$, the law of the propagation of errors assumes the form

$$s_f^2 = s_x^2 + a^2 s_y^2$$

It is found that considerable simplification results if the variability in ballistic performance is ascribed to the parameters c, g, and h with the parameters n, m, and q assumed constant. This is the same assumption used to simplify the treatment of the effect of temperature. On this basis one derives from the equation $p = (K/g)^{1/m}$ the variance of chamber pressure at constant area ratio as

$$(s_p^2)_K = (1/m)^2 s_g^2 \qquad [28]$$

The variance of g is obtainable from a quality-control chart of the quantity K/p^m; hence $(s_p^2)_K$ can be calculated in two ways, either directly from a series of firings at constant area ratio or indirectly from a quality-control chart based on firings at varying K's by the application of Equation [28].

From the equation $C_w = hp^q$ one can obtain

$$(s_{C_w}^2)_p = s_h^2 \qquad [29]$$

and

$$(s_{C_w}^2)_K = s_h^2 + q^2(s_p^2)_K$$

$$= s_h^2 + (q/m)^2 s_g^2 \qquad [30]$$

or

$$(s_{C_w}^2)_K = (s_{C_w}^2)_p + q^2(s_p^2)_K \qquad [31]$$

The variance of h is obtainable from a quality-control chart of the quantity C_w/p^q; hence $(s_{C_w}^2)_K$ can be obtained either directly from a series of firings at constant area ratio or indirectly from quality-control charts based on firings at varying K's by the application of Equation [30].

From the equation $r = cp^n$ one can obtain

$$(s_r^2)_p = s_c^2 \qquad [32]$$

and

$$(s_r^2)_K = s_c^2 + n^2(s_p^2)_K$$

$$= s_c^2 + (n/m)^2 s_g^2 \qquad [33]$$

or

$$(s_r^2)_K = (s_r^2)_p + n^2(s_p^2)_K \qquad [34]$$

The variance of c is obtainable from a quality-control chart of the quantity r/p^n; hence $(s_r^2)_K$ can be obtained either directly from a series of firings at constant area ratio or indirectly from quality-control charts based on firings at varying K's by the application of Equation [33].

The pertinent relationships are summarized in Table 2.

TABLE 2 SUMMARY OF VARIANCES

$(s_{Cw}^2)_p =$	$s_h^2 =$ (s_h from control chart of C_w/p^q)	$(s_{Cw}^2)_K - q^2(s_p^2)_K$
$(s_r^2)_p =$	$s_c^2 =$ (s_c from control chart of r/p^n)	$(s_r^2)_K - n^2(s_p^2)_K$
$(s_p^2)_K =$ (s_p from constant K firings)	$(1/m)^2 s_g^2 =$ (s_g from control chart of K/p^m)	$(s_r^2)_K + (s_{Cw}^2)_K = [(s_r^2)_p + (s_{Cw}^2)_p]/(1 - n^2 - q^2)$
$(s_{Cw}^2)_K =$ (s_{Cw} from constant K firings)	$s_h^2 + (q/m)^2 s_g^2 =$	$(s_{Cw}^2)_p + q^2(s_p^2)_K$
$(s_r^2)_K =$ (s_r from constant K firings)	$s_c^2 + (n/m)^2 s_g^2 =$	$(s_r^2)_p + n^2(s_p^2)_K$

By their use the standard deviations of a given parameter can be estimated, in the absence of direct data, from standard deviations known for other parameters.

Applications

The equations developed in the two preceding sections may be more easily understood when applied to specific test results. For this purpose, data from 57 static firings obtained during

TABLE 3 PERFORMANCE DATA AT −40 F

Test no.	Chamber pressure, p 1000 lbf in.^{2}	Burning rate, r in sec^{-1}	Area ratio, K	Mass flow coefficient, C_w $\frac{\text{lbm}}{\text{1000 lbf sec}}$	r/p^n $(= c)$	K/p^m $(= g)$	C_w/p^q $(= h)$
1	0.936	0.647	189.6	9.04	0.679	192.5	9.02
2	0.949	0.656	189.8	9.08	0.682	192.1	9.07
3	0.945	0.659	189.5	9.11	0.687	192.0	9.10
4	0.937	0.644	190.2	9.00	0.676	193.1	8.98
5	0.964	0.676	190.1	9.13	0.695	191.7	9.12
6	0.952	0.662	189.7	9.04	0.687	191.9	9.03
7	1.018	0.666	189.3	—	0.657	188.5	—
8	0.980	0.668	190.3	8.94	0.678	191.2	8.93
9	0.965	0.665	190.3	9.00	0.683	191.9	8.99
10	1.032	0.706	189.7	8.97	0.690	188.3	8.98
11	1.075	0.723	190.4	8.84	0.685	187.2	8.86
12	0.975	0.664	190.1	8.88	0.677	191.2	8.87
13	0.977	0.656	190.4	8.79	0.667	191.4	8.78
14	0.987	0.663	190.1	8.83	0.669	190.7	8.83
15	0.974	0.659	189.9	8.88	0.672	191.1	8.87
16	0.991	0.665	190.5	8.77	0.669	190.9	8.77
17	0.992	0.660	190.5	8.74	0.664	190.9	8.74
$\bar{x}$	0.979	0.667	190.0	8.94	0.678	191.0	8.93
s_x, %	3.68	2.94	0.19	1.40	1.50	0.82	1.35

TABLE 4 PERFORMANCE DATA AT +60 F

Test no.	Chamber pressure, p 1000 lbf in.$^{-2}$	Burning rate, r in sec^{-1}	Area ratio, K	Mass flow coefficient, C_w $\frac{\text{lbm}}{\text{1000 lbf sec}}$	r/p^n $(= c)$	K/p^m $(= g)$	C_w/p^q $(= h)$
18	1.180	0.816	190.0	9.05	0.772	182.9	9.09
19	1.227	0.819	190.3	8.71	0.704	181.5	8.76
20	1.262	0.852	190.2	8.82	0.717	180.3	8.88
21	1.234	0.830	189.3	8.82	0.710	180.3	8.87
22	1.181	0.810	189.7	8.92	0.716	182.5	8.96
23	1.220	0.831	189.7	8.89	0.717	181.5	8.84
24	1.219	0.844	190.2	9.02	0.729	181.7	9.07
25	1.206	0.828	189.7	8.89	0.721	181.7	8.94
26	1.254	0.822	189.6	8.51	0.695	179.9	8.57
27	1.214	0.832	190.0	8.99	0.721	181.7	9.04
28	1.339	0.918	190.2	8.97	0.740	177.8	9.04
29	1.355	0.913	190.4	8.81	0.729	177.5	8.89
30	1.327	0.906	189.6	8.91	0.735	177.6	8.98
31	1.366	0.926	190.3	8.86	0.735	177.1	8.94
32	1.365	0.909	190.3	8.72	0.722	177.1	8.80
33	1.242	0.836	190.3	8.79	0.712	181.0	8.85
34	1.271	0.835	189.8	8.65	0.699	179.6	8.71
35	1.233	0.831	190.2	8.83	0.712	181.2	8.88
36	1.250	0.823	189.8	8.64	0.698	180.3	8.70
37	1.221	0.821	190.0	8.81	0.708	181.4	8.86
$\bar{x}$	1.258	0.850	190.0	8.83	0.717	180.2	8.89
s_x, %	4.74	4.64	0.16	1.54	1.77	1.00	1.50

TABLE 5 PERFORMANCE DATA AT +140 F

Test no.	Chamber pressure, p 1000 lbf in.$^{-2}$	Burning rate, r in sec^{-1}	Area ratio, K	Mass flow coefficient, C_w $\frac{\text{lbm}}{\text{1000 lbf sec}}$	r/p^n $(= c)$	K/p^m $(= g)$	C_w/p^q $(= h)$
38	1.556	1.070	190.0	9.02	0.771	171.6	9.14
39	1.604	1.052	189.9	8.54	0.742	170.3	8.66
40	1.641	1.075	189.7	8.57	0.745	169.2	8.69
41	1.594	1.075	189.8	8.84	0.761	170.4	8.96
42	1.588	1.084	189.9	8.83	0.770	170.7	8.95
43	1.609	1.071	189.9	8.66	0.753	170.1	8.78
44	1.597	1.058	189.3	8.63	0.748	169.9	8.75
45	1.581	1.070	189.6	8.79	0.762	170.6	8.91
46	1.607	1.097	190.1	8.92	0.772	170.4	9.04
47	1.592	1.089	189.7	8.84	0.772	170.4	8.96
48	1.672	1.079	189.6	8.42	0.738	168.4	8.55
49	1.590	1.088	190.1	8.97	0.772	170.8	9.09
50	1.605	1.094	189.4	8.84	0.771	169.8	8.96
51	1.595	1.088	189.8	8.92	0.770	170.4	9.04
52	1.732	1.169	190.1	8.82	0.779	167.4	8.96
53	1.785	1.199	190.2	8.93	0.781	166.4	9.08
54	1.769	1.199	189.9	8.85	0.786	166.5	9.00
55	1.755	1.183	190.0	8.76	0.780	166.9	8.90
56	1.744	1.191	190.2	8.93	0.789	167.3	9.08
57	1.555	1.068	190.0	8.95	0.770	171.6	9.07
$\bar{x}$	1.639	1.105	189.9	8.80	0.767	169.5	8.93
s_x, %	4.58	4.61	0.13	1.82	1.90	0.99	1.82

testing of a composite propellant based on potassium perchlorate in a neutral-burning configuration will be utilized (3). Erosive burning of this propellant was negligible under the conditions used. All firings were made at a nearly constant area ratio, $K = 190$, with 17 firings at a propellant temperature of −40 F, 20 firings at +60 F, and 20 firings at +140 F. The observed data, segregated according to temperature, are given in Tables 3, 4, and 5.

The density of the propellant used was 0.0681 lbm in.$^{-3}$. The values of the exponents in Equations [1], [4], and [7] were known from previous experimental work to be $n = 0.740$, $q = -0.029$, and $m = 0.231$. These values are self-consistent inasmuch as they satisfy Equation [9].

Values of the parameter c, h, and g were calculated for each firing from Equations [1], [4], and [7], respectively, and the results[4] are also tabulated in Tables 3, 4, and 5. The average values for each of the three temperatures are summarized in Table 6. A check of the consistency of the data is afforded by calculating g at each temperature from Equation [8]

$$\text{at } -40 \text{ F } g = \frac{8.93}{0.678\,(0.0681 - 0.0005)} = 194.9$$

$$\text{at } 60 \text{ F } g = \frac{8.89}{0.717\,(0.0681 - 0.0005)} = 183.4$$

$$\text{at } 140 \text{ F } g = \frac{8.93}{0.767\,(0.00681 - 0.0005)} = 172.2$$

Agreement with the observed values of g in Table 6 is considered satisfactory.

[4] Estimated standard deviations are computed by the formula

$$s_x = \sqrt{\frac{\Sigma(x_i - \bar{x})^2}{N - 1}}$$

at constant temperature, namely, c, g, h, n, m, and q. However, in practice it is found that a certain amount of variability in these quantities from one sample of propellant to another must be taken into account. Inasmuch as reproducibility in propellant performance is of utmost importance in manufacturing rockets economically, a measure of variability is necessary. The variability is most conveniently expressed in terms of the estimated standard deviations of the ballistic parameters, and the relations between the various estimated standard deviations can be found by utilizing the law of propagation of error for uncorrelated variables. This law states that if

$$f = f(x, y, z)$$

then

$$\sigma_f^2 = \left(\frac{\partial f}{\partial x}\right)^2 \sigma_x^2 + \left(\frac{\partial f}{\partial y}\right)^2 \sigma_y^2 + \left(\frac{\partial f}{\partial z}\right)^2 \sigma_z^2$$

In the particular case where

$$f = xy^a$$

and when the standard deviations are expressed in fractional or percentage values, symbolized by $s_x = 100\sigma_x/x$, the law of the propagation of errors assumes the form

$$s_f^2 = s_x^2 + a^2 s_y^2$$

It is found that considerable simplification results if the variability in ballistic performance is ascribed to the parameters c, g, and h with the parameters n, m, and q assumed constant. This is the same assumption used to simplify the treatment of the effect of temperature. On this basis one derives from the equation $p = (K/g)^{1/m}$ the variance of chamber pressure at constant area ratio as

$$(s_p^2)_K = (1/m)^2 s_g^2 \qquad [28]$$

The variance of g is obtainable from a quality-control chart of the quantity K/p^m; hence $(s_p^2)_K$ can be calculated in two ways, either directly from a series of firings at constant area ratio or indirectly from a quality-control chart based on firings at varying K's by the application of Equation [28].

From the equation $C_w = hp^q$ one can obtain

$$(s_{C_w}^2)_p = s_h^2 \qquad [29]$$

and

$$(s_{C_w}^2)_K = s_h^2 + q^2(s_p^2)_K$$

$$= s_h^2 + (q/m)^2 s_g^2 \qquad [30]$$

or

$$(s_{C_w}^2)_K = (s_{C_w}^2)_p + q^2(s_p^2)_K \qquad [31]$$

The variance of h is obtainable from a quality-control chart of the quantity C_w/p^q; hence $(s_{C_w}^2)_K$ can be obtained either directly from a series of firings at constant area ratio or indirectly from quality-control charts based on firings at varying K's by the application of Equation [30].

From the equation $r = cp^n$ one can obtain

$$(s_r^2)_p = s_c^2 \qquad [32]$$

and

$$(s_r^2)_K = s_c^2 + n^2(s_p^2)_K$$

$$= s_c^2 + (n/m)^2 s_g^2 \qquad [33]$$

or

$$(s_r^2)_K = (s_r^2)_p + n^2(s_p^2)_K \qquad [34]$$

The variance of c is obtainable from a quality-control chart of the quantity r/p^n; hence $(s_r^2)_K$ can be obtained either directly from a series of firings at constant area ratio or indirectly from quality-control charts based on firings at varying K's by the application of Equation [33].

The pertinent relationships are summarized in Table 2.

TABLE 2 SUMMARY OF VARIANCES

$(s_{Cw}^2)_p =$	$s_h^2 =$ (s_h from control chart of C_w/p^q)	$(s_{Cw}^2)_K - q^2(s_p^2)_K$
$(s_r^2)_p =$	$s_c^2 =$ (s_c from control chart of r/p^n)	$(s_r^2)_K - n^2(s_p^2)_K$
$(s_p^2)_K =$ (s_p from constant K firings)	$(1/m)^2 s_g^2 =$ (s_g from control chart of K/p^m)	$(s_r^2)_K + (s_{Cw}^2)_K = [(s_r^2)_p + (s_{Cw}^2)_p]/(1 - n^2 - q^2)$
$(s_{Cw}^2)_K =$ (s_{Cw} from constant K firings)	$s_h^2 + (q/m)^2 s_g^2 =$	$(s_{Cw}^2)_p + q^2(s_p^2)_K$
$(s_r^2)_K =$ (s_r from constant K firings)	$s_c^2 + (n/m)^2 s_g^2 =$	$(s_r^2)_p + n^2(s_p^2)_K$

By their use the standard deviations of a given parameter can be estimated, in the absence of direct data, from standard deviations known for other parameters.

Applications

The equations developed in the two preceding sections may be more easily understood when applied to specific test results. For this purpose, data from 57 static firings obtained during

TABLE 3 PERFORMANCE DATA AT −40 F

Test no.	Chamber pressure, p 1000 lbf in.^{2}	Burning rate, r in sec^{-1}	Area ratio, K	Mass flow coefficient, C_w lbm / 1000 lbf sec	r/p^n $(= c)$	K/p^m $(= g)$	C_w/p^q $(= h)$
1	0.936	0.647	189.6	9.04	0.679	192.5	9.02
2	0.949	0.656	189.8	9.08	0.682	192.1	9.07
3	0.945	0.659	189.5	9.11	0.687	192.0	9.10
4	0.937	0.644	190.2	9.00	0.676	193.1	8.98
5	0.964	0.676	190.1	9.13	0.695	191.7	9.12
6	0.952	0.662	189.7	9.04	0.687	191.9	9.03
7	1.018	0.666	189.3	—	0.657	188.5	—
8	0.980	0.668	190.3	8.94	0.678	191.2	8.93
9	0.965	0.665	190.3	9.00	0.683	191.9	8.99
10	1.032	0.706	189.7	8.97	0.690	188.3	8.98
11	1.075	0.723	190.4	8.84	0.685	187.2	8.86
12	0.975	0.664	190.1	8.88	0.677	191.2	8.87
13	0.977	0.656	190.4	8.79	0.667	191.4	8.78
14	0.987	0.663	190.1	8.83	0.669	190.7	8.83
15	0.974	0.659	189.9	8.88	0.672	191.1	8.87
16	0.991	0.665	190.5	8.77	0.669	190.9	8.77
17	0.992	0.660	190.5	8.74	0.664	190.9	8.74
$\bar{x}$	0.979	0.667	190.0	8.94	0.678	191.0	8.93
s_x, %	3.68	2.94	0.19	1.40	1.50	0.82	1.35

TABLE 4 PERFORMANCE DATA AT +60 F

Test no.	Chamber pressure, p 1000 lbf in.$^{-2}$	Burning rate, r in sec^{-1}	Area ratio, K	Mass flow coefficient, C_w $\frac{\text{lbm}}{\text{1000 lbf sec}}$	r/p^n ($= c$)	K/p^m ($= g$)	C_w/p^q ($= h$)
18	1.180	0.816	190.0	9.05	0.772	182.9	9.09
19	1.227	0.819	190.3	8.71	0.704	181.5	8.76
20	1.262	0.852	190.2	8.82	0.717	180.3	8.88
21	1.234	0.830	189.3	8.82	0.710	180.3	8.87
22	1.181	0.810	189.7	8.92	0.716	182.5	8.96
23	1.220	0.831	189.7	8.89	0.717	181.5	8.84
24	1.219	0.844	190.2	9.02	0.729	181.7	9.07
25	1.206	0.828	189.7	8.89	0.721	181.7	8.94
26	1.254	0.822	189.6	8.51	0.695	179.9	8.57
27	1.214	0.832	190.0	8.99	0.721	181.7	9.04
28	1.339	0.918	190.2	8.97	0.740	177.8	9.04
29	1.355	0.913	190.4	8.81	0.729	177.5	8.89
30	1.327	0.906	189.6	8.91	0.735	177.6	8.98
31	1.366	0.926	190.3	8.86	0.735	177.1	8.94
32	1.365	0.909	190.3	8.72	0.722	177.1	8.80
33	1.242	0.836	190.3	8.79	0.712	181.0	8.85
34	1.271	0.835	189.8	8.65	0.699	179.6	8.71
35	1.233	0.831	190.2	8.83	0.712	181.2	8.88
36	1.250	0.823	189.8	8.64	0.698	180.3	8.70
37	1.221	0.821	190.0	8.81	0.708	181.4	8.86
$\bar{x}$	1.258	0.850	190.0	8.83	0.717	180.2	8.89
s_x, %	4.74	4.64	0.16	1.54	1.77	1.00	1.50

TABLE 5 PERFORMANCE DATA AT +140 F

Test no.	Chamber pressure, p 1000 lbf in.$^{-2}$	Burning rate, r in sec^{-1}	Area ratio, K	Mass flow coefficient, C_w $\frac{\text{lbm}}{\text{1000 lbf sec}}$	r/p^n ($= c$)	K/p^m ($= g$)	C_w/p^q ($= h$)
38	1.556	1.070	190.0	9.02	0.771	171.6	9.14
39	1.604	1.052	189.9	8.54	0.742	170.3	8.66
40	1.641	1.075	189.7	8.57	0.745	169.2	8.69
41	1.594	1.075	189.8	8.84	0.761	170.4	8.96
42	1.588	1.084	189.9	8.83	0.770	170.7	8.95
43	1.609	1.071	189.9	8.66	0.753	170.1	8.78
44	1.597	1.058	189.3	8.63	0.748	169.9	8.75
45	1.581	1.070	189.6	8.79	0.762	170.6	8.91
46	1.607	1.097	190.1	8.92	0.772	170.4	9.04
47	1.592	1.089	189.7	8.84	0.772	170.4	8.96
48	1.672	1.079	189.6	8.42	0.738	168.4	8.55
49	1.590	1.088	190.1	8.97	0.772	170.8	9.09
50	1.605	1.094	189.4	8.84	0.771	169.8	8.96
51	1.595	1.088	189.8	8.92	0.770	170.4	9.04
52	1.732	1.169	190.1	8.82	0.779	167.4	8.96
53	1.785	1.199	190.2	8.93	0.781	166.4	9.08
54	1.769	1.199	189.9	8.85	0.786	166.5	9.00
55	1.755	1.183	190.0	8.76	0.780	166.9	8.90
56	1.744	1.191	190.2	8.93	0.789	167.3	9.08
57	1.555	1.068	190.0	8.95	0.770	171.6	9.07
$\bar{x}$	1.639	1.105	189.9	8.80	0.767	169.5	8.93
s_x, %	4.58	4.61	0.13	1.82	1.90	0.99	1.82

testing of a composite propellant based on potassium perchlorate in a neutral-burning configuration will be utilized (3). Erosive burning of this propellant was negligible under the conditions used. All firings were made at a nearly constant area ratio, $K = 190$, with 17 firings at a propellant temperature of -40 F, 20 firings at $+60$ F, and 20 firings at $+140$ F. The observed data, segregated according to temperature, are given in Tables 3, 4, and 5.

The density of the propellant used was 0.0681 lbm in.$^{-3}$. The values of the exponents in Equations [1], [4], and [7] were known from previous experimental work to be $n = 0.740$, $q = -0.029$, and $m = 0.231$. These values are self-consistent inasmuch as they satisfy Equation [9].

Values of the parameter c, h, and g were calculated for each firing from Equations [1], [4], and [7], respectively, and the results[4] are also tabulated in Tables 3, 4, and 5. The average values for each of the three temperatures are summarized in Table 6. A check of the consistency of the data is afforded by calculating g at each temperature from Equation [8]

$$\text{at } -40 \text{ F } g = \frac{8.93}{0.678\,(0.0681 - 0.0005)} = 194.9$$

$$\text{at } 60 \text{ F } g = \frac{8.89}{0.717\,(0.0681 - 0.0005)} = 183.4$$

$$\text{at } 140 \text{ F } g = \frac{8.93}{0.767\,(0.00681 - 0.0005)} = 172.2$$

Agreement with the observed values of g in Table 6 is considered satisfactory.

[4] Estimated standard deviations are computed by the formula

$$s_x = \sqrt{\frac{\Sigma(x_i - \bar{x})^2}{N - 1}}$$

TABLE 6 SUMMARY OF TEMPERATURE COEFFICIENTS

Ballistic property	Absolute values −40 F	+60 F	+140 F	Symbol	Relative temperature coefficients −40 F to +60 F	+60 F to +140 F
c	0.678	0.717	0.767	u	+0.00055	+0.00087
h	8.93	8.89	8.93	v	0.00000	0.00000
g	191.0	180.2	169.5	w	−0.00058	−0.00077
p, $K = 190.0$	0.979	1.258	1.639	π_p	+0.00251	+0.00331
r, $K = 190.0$	0.667	0.850	1.105	π_r	+0.00243	+0.00327
C_w, $K = 190.0$	8.94	8.83	8.80	π_C	−0.00008	−0.00004

Equation	Theoretical relation	−40 F to +60 F	+60 F to +140 F
[12]	$w = v - u$	$-0.00058 = -0.00055$	$-0.00077 = -0.00087$
[15]	$\pi_p = -w/m$	$0.00251 = 0.00251$	$0.00331 = 0.00333$
[20]	$\pi_C = v - (q/m)w$	$-0.00008 = -0.00007$	$-0.00004 = -0.00009$
[25]	$\pi_r = u - (n/m)w$	$0.00243 = 0.00241$	$0.00327 = 0.00334$
[27]	$\pi_p = \pi_r - \pi_C$	$0.00251 = 0.00251$	$0.00331 = 0.00331$

TABLE 7 SUMMARY OF VARIANCES

	−40 F $\bar{x}$	s, %	s^2	+60 F $\bar{x}$	s, %	s^2	+140 F $\bar{x}$	s, %	s^2
p	0.979	3.68	13.54	1.258	4.74	22.47	1.639	4.58	20.98
r	0.667	2.94	8.64	0.850	4.64	21.53	1.105	4.61	21.25
C_w	8.94	1.40	1.96	8.83	1.54	2.37	8.80	1.82	3.31
c	0.678	1.50	2.25	0.717	1.77	3.13	0.767	1.90	3.61
g	191.0	0.82	0.67	180.2	1.00	1.00	169.5	0.99	0.98
h	8.93	1.35	1.82	8.89	1.50	2.25	0.893	1.82	3.31

Equation	Theoretical relation	−40 F	+60 F	+140 F
[28]	$(s_p^2)_K = (1/m)^2 s_g^2$	$13.5 \cong 12.5$	$22.5 \cong 18.7$	$21.0 \cong 18.4$
[30]	$(s_{Cw}^2) = s_h^2 + (q/m)^2 s_g^2$	$1.96 \cong 1.83$	$2.37 \cong 2.26$	$3.31 \cong 3.32$
[33]	$(s_r^2)_K = s_c^2 + (n/m)^2 s_g^2$	$8.64 \cong 9.13$	$21.5 \cong 13.4$	$21.25 = 13.67$

Plots against propellant temperature of log c, log h, and log g from Table 6 are approximately linear from −40 to 140 F which indicates that Equations [2], [5], and [10] are reasonable representations of the effect of propellant temperature. However, somewhat greater accuracy is obtained by using different slopes above and below 60 F. Values of the slopes u, v, and w fitting the observed data in each temperature range are summarized in Table 6. The value of $u = 0.00055$, for example, means that the propellant burning rate below 60 F increases 0.055 per cent per °F at a given chamber pressure.

Similar plots against propellant temperature of log p, log r, and log C_w from Table 6 yield values of the slopes π_p, π_r, and π_C; these are summarized in the table. The value of $\pi_r = 0.00243$, for example, means that the propellant burning rate below 60 F increases 0.243 per cent per °F at a given constant area ratio K. It is evident from the calculations summarized at the bottom of Table 6 that the various relations among u, v, w, π_p, π_r, and π_C are satisfied within experimental error.

Reproducibility of the propellant can be judged from the estimated standard deviations given at the bottom of Tables 3, 4, and 5 and summarized in Table 7. For example, the standard deviation of the burning rate at constant area ratio and at 60 F is 4.64 per cent. This is much larger than the standard deviation of the constant c in the burning-rate equation which is only 1.77 per cent. The difference is due to the magnifying effect of the burning-rate exponent implicit in Equation [13].

It is evident from the calculations summarized at the bottom of Table 7 that the relations among the various standard deviations are satisfied sufficiently well to justify the assumptions made in their deviations. Even better agreement could have been obtained by taking account of the fact that the area ratio K in the tests reported was not absolutely constant.

In view of this agreement with experimental data, it can be concluded that the equations given in the present paper will be useful in correlating data for other propellants and in estimating parameters often needed in rocket design when they have not been directly measured.

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High-Frequency Combustion Instability in Solid Propellant Rockets. Part 1[1]

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A theory of unstable high-frequency oscillations in solid propellant rockets is advanced with the mechanism of self-excitation based on the following simplified model. Both the rate of primary decomposition of solid propellant element and the rate of activation of the intermediate products during the time lag are assumed to depend upon the gas oscillations. Both interactions are expressed in terms of the instantaneous pressure according to power law with exponent n and m respectively. The dependence of the rate of decomposition on drifting velocity is shown to be not of fundamental importance in determining the stability of oscillations in the combustion chamber. Analysis shows that a simplified over-all pressure index of interaction $S = m - (n/2)$ must be bigger than certain minimum value if unstable oscillation is to be possible. Unstable oscillation further requires the time lag to be in proper range. The most unstable mode is shown to be the fundamental spiral mode, and its tangential component with exactly the acoustical frequency is the one which is most easily observed at finite magnitudes because of its rapid amplification. Several configurations of the propellant grain with radial burning surfaces have been investigated. Rod grain becomes more stable while other grain shapes become less stable in the course of firing. Rod grain is the most stable while tubular grain is the most unstable configuration. The stabilizing effect of inserting a nonburning solid rod into tubular grain is verified by the theory. Several other aspects have also been treated. Solid rocket with end burning grain also becomes less stable in the course of operation. Both the spiral and the longitudinal modes in rocket with end-burning grain are briefly investigated.

Nomenclature

$\dot{m}_\xi$ = rate of decomposition of the solid propellant per unit area of the burning surface

$\dot{m}_b$ = rate of generation of the final combustion products per unit area of the burning surface

τ, $\bar{\tau}$ = dimensionless instantaneous and the steady-state values of the time lag that the intermediate gases take before they are completely burned

m = pressure index of interaction between the combustion reactions in gaseous phase and the oscillations in the burned gas

n = pressure index of interaction between the decomposition reactions of the solid propellant and the oscillations in the burned gas

$S = m - n/2$ = over-all pressure index of interaction

γ = specific heat ratio

x, r, and θ = spatial cylindrical coordinates with length nondimensionalized by the radius of the cylindrical cavity of the combustion space

λ = dimensionless length of the propellant grain

u, v, and w = instantaneous velocity components in x, r, and θ direction nondimensionalized by speed of sound in stagnant burned gas

p and ρ = instantaneous gas pressure and density nondimensionalized by the pressure and density of stagnant burned gas, respectively

μ, ν, and χ = time independent parts of small perturbations over the dimensionless steady-state velocity components u, v, and w, respectively

φ and δ = time independent parts of small perturbations over the dimensionless steady-state values of pressure and density, respectively

$i\alpha = i\omega - k$ = root of the characteristic equation with the dimensionless time as the independent variable

Ⓗ = reference or characteristic time for the sound wave to travel a distance equal to the radius of the cylindrical cavity in stagnant burned gas

ω = dimensionless angular frequency

k = dimensionless damping coefficient

ξ, η, and β = characteristic constants in the solution of the wave equation that indicates the mode of oscillation in the axial, radial, and the circumferential directions, respectively

ξ_r and ξ_i = real and imaginary parts of ξ

$\bar{u}_e$ = dimensionless mean gas velocity at axial exit of propellant grain

$A = A_r + iA_i$ = ratio of fractional variation of velocity to that of density at $x = \lambda$

e = ratio of the distance from the axial exit $x = \lambda$ to sonic throat of the nozzle to the length of the propellant grain

Ω = reduced frequency parameter in the effective nozzle

v_1 = velocity of the burned gas normal to and at the burning surface

κ = parameter of radial dimensions and β as defined in Equation [20]

σ $= -\dfrac{\bar{u}_e}{2\lambda\bar{v}_1}\kappa$

$\zeta = \omega\dfrac{A_i A_r \bar{u}_e}{2\lambda|\bar{v}_1|}$ = parameter of boundary condition at axial exit

a = dimensionless radius of the second solid boundary in radial direction

B $= 1 - \gamma m + \gamma(m - n)\exp(-i\alpha\bar{\tau})$ = parameter of boundary condition on burning surface

Subscripts x, r, and θ = partial differentiation with respect to the independent variable indicated

Superscripts $^{(0)}$ and $^{(1)}$ = order of successive approximation

$\Re$ and $\Im$ = real and imaginary part of the quantity in the bracket

$|\quad|$ indicates the absolute magnitude of the complex quantity

Superscript * means the quantity is dimensional

$^-$ over letter indicates mean or steady-state quantity

Introduction

A SOLID propellant rocket is schematically represented by a duct of uniform cross-sectional area followed by a de Laval nozzle with supersonic exit. Combustion gases are generated from the burning surface of the solid propellant in the combustion chamber and leave the combustion chamber through the de Laval nozzle under the combustion chamber

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pressure p_c. From the consideration of mass continuity, the criterion of smooth operation is given by $n < 1$, where n is the pressure index in the conventional burning law $\dot{m}_b \sim p_c^n$. However, abnormal pressure peaks have frequently been observed in rockets using propellants with $n < 1$. This abnormal phenomenon has not been satisfactorily explained but is generally considered as the manifestation of some unstable phenomenon due to the variation of the rate of combustion gas generation created by disturbances in the combustion chamber. It has been found experimentally (1)[3] that high-frequency oscillations of finite amplitudes are associated with abnormal pressure peaks and that the frequencies correspond closely with the characteristic frequencies of the transversal acoustical modes of oscillations in the combustion system. The stability of high-frequency transversal modes of oscillations in solid rockets has been investigated in (2). However, conclusions concerning the relative stability of systems with different values of the governing parameters have not been deduced. Hence, it is not possible to tell by comparing with experiments whether the mechanism of self-excitation and the uncertain boundary condition as postulated in (2) can represent the principal features of the problem.

It is well known that instability of high-frequency oscillations associated with abnormal pressure peaks is found mostly in rockets using tubular or perforated cylindrical grain with interior burning surface. Instability has been demonstrated in rockets using rod-in-tube grain (1), but it has never been demonstrated in rockets using rod grain with exterior burning surface. It is also known that instability in rockets using tubular grain can often be suppressed by installing a nonburning rod in the perforation. The trend of the relative stability of small disturbances in these rocket systems is rather clear. Furthermore, the fact that abnormal pressure peak is always preceded by a period of smooth operation indicates that the solid rocket tends to destabilize itself during the course of operation. Any stability theory based upon some postulated mechanism of self-excitation must be able to verify all these qualitative trends before it can be considered as satisfactory. Then other deductions from this theory can probably lead to the ways of avoiding or suppressing the undesirable instability.

In the present investigation, the stability of high-frequency small periodic disturbances superposed on steady-state flow is studied. The magnitude of the perturbations is assumed to be so small that terms involving squares or products of the perturbations are negligible as compared to terms involving the perturbations only to the first power. The large amplitude oscillations as observed in rough burning rockets are considered to be the result of amplification of the unstable small oscillations, because it is not likely that oscillations of finite magnitudes are introduced into the flow system from outside directly. If the small oscillations are unstable the amplitudes of these small disturbances will grow with time. Thus new damping mechanisms and new exciting mechanisms of nonlinear character will gradually come into play and finally become predominant in limiting the magnitudes of the unstable oscillations. If the small oscillations are stable in the flow system, these oscillations will die out and the nonlinear mechanisms have no chance to play its role. Therefore, a linearized analysis for the stability of small perturbations can reveal whether oscillations of finite magnitudes are possible in a given system even if such linearized analysis cannot predict the behavior or properties of the oscillations of finite magnitudes.

When unstable spiral modes of oscillations grow in a solid rocket, the oscillation of the circumferential velocity component, as explained in appendix 2, brings about an effective nonlinear mechanism of excitation which does not exist when unstable nonspiral modes grow in magnitude. This additional exciting mechanism results in bigger increase in the mean burning rate which provides more energy for the rapid amplification of the unstable disturbances. The unstable spiral mode will therefore amplify faster and will more likely be observed at finite magnitudes than the unstable nonspiral mode. The abnormal increase in the mean burning rate and the mean chamber pressure associated with spiral modes of oscillations at finite magnitudes is considered as a property of the oscillations at finite magnitudes. It is therefore not considered to be important in building up the simplified model of interaction between the combustion processes and the oscillations in burned gas for linearized stability analysis.

The simplified model of interaction between the combustion processes and the oscillations in gaseous phase has to be postulated. Owing to the lack of good understanding of the kinetics of the combustion processes of solid propellants, any assumptions about the variation of the rate of combustion gas generation created by disturbances in gaseous phase are more or less speculative. We are trying to build up a simplified model that will lead to analytical results that agree in qualitative trend with experimental facts. If this can be achieved, this simplified model will be expected to have included the principal features of the problem.

Mechanism of Self-Excitation

The combustion processes on the solid surface are not well understood, but can be described qualitatively as consisting of two major stages. Solid propellant elements near the surface which are energized either by heat transfer processes or by direct impingement of the activated species from neighboring gas will undergo a primary decomposition into intermediate gases. This primary decomposition is either endothermic or slightly exothermic. The intermediate gas is further activated by the neighboring burned gas during a certain time interval τ, called the time lag, and then undergoes a series of complicated chemical reactions at a large but finite rate toward final combustion products. Practically all the available chemical energy of the propellant is transformed into thermal energy through the combustion reactions in gaseous phase. The kinetics of both primary decomposition and combustion reactions are extremely complicated. Hence, we shall assume that the primary decomposition generates the intermediate gases at a rate $\dot{m}_i(t)$ and that the intermediate gases generated at the instant t will burn instantaneously and completely at the instant $t + \tau$. In unsteady-state operation both $\dot{m}_i$ and τ will vary. The instantaneous rate of burned gas generation can be shown (4) to be

$$\dot{m}_b(t) = \left(1 - \frac{d\tau}{dt}\right) \dot{m}_i(t - \tau) \qquad [1]$$

The time lag τ of the intermediate gas is affected by the oscillations of pressure and temperature of the burned gas. Since the small temperature and the small pressure oscillations in burned gas are correlated in the simplified model of instantaneous combustion, the variation of the time lag can be expressed in terms of pressure only. We shall express the relation as

$$\int_{t-\tau}^{t} p^m(t')\, dt' = \text{const} \qquad [2]$$

where $p(t')$ is the instantaneous pressure of the burned gas acting on the intermediate gas at the instant t' (3, 4). The constant in Equation [2] may be roughly interpreted as the total amount of activation energy that an average combustible gaseous element will absorb before its complete combustion, and $p^m(t')$ is the local rate of the activation process at the instant t'.

In steady-state operation, τ is constant and $\dot{m}_b$ is equal to $\dot{m}_i$. Therefore, to be consistent with the steady-state burning law, it will be assumed that the instantaneous rate of de-

[3] Numbers in parentheses refer to the References on page 32.

composition $\dot{m}_i(t)$ is related to the instantaneous gas pressure and drifting velocity in the following manner.

$$\dot{m}_i(t) = \text{const} \cdot p^n(1 + bV) \qquad [3]$$

where V is the velocity component of the burned gas parallel to the burning surface, i.e., the drifting velocity.

Substituting Equations [2] and [3] in [1], we obtain the ratio of the instantaneous rate of burned gas generation and the steady-state burning rate as

$$\frac{\dot{m}_b(t)}{\bar{\dot{m}}_b} = \left(1 - \frac{d\tau}{dt}\right) \frac{p^n(t-\tau)}{\bar{p}^n} \cdot \frac{1 + bV}{1 + b\bar{V}} = \frac{p^m(t)}{[p(t - \bar{\tau})]^{m-n}} \cdot \frac{1}{\bar{p}^n} \cdot \frac{1 + bV}{1 + b\bar{V}} \qquad [4]$$

where τ is replaced by $\bar{\tau}$ within the framework of small perturbation analysis. The variation of the burning rate can be recognized as consisting of two parts: a pressure dependent part, and a velocity dependent part. It will be shown in appendix 2 that the dependence of the burning rate on the drifting velocity alone is not likely to excite instability despite the fact that it is an important nonlinear mechanism in increasing the mean burning rate and the mean chamber pressure. The present investigation is, therefore, made on the opposite extreme assumption that the instantaneous burning rate depends on pressure only. Thus

$$\frac{\dot{m}_b(t)}{\bar{\dot{m}}_b} = \frac{p^m(t)}{[p(t-\tau)]^{m-n}} \cdot \frac{1}{\bar{p}^n(t)} \qquad [5]$$

where m and n are the pressure indices of interaction of the gaseous phase reactions and the solid phase reactions, respectively. Since these quantities m and n are the average interaction indices between the rates of combustion processes and oscillations in burned gas, their values for different propellants have to be determined by experiments. At the present stage of our knowledge about the kinetics of combustion, it is not possible to determine m and n theoretically with reasonable accuracy. The value of n can probably be taken as the value of the pressure index in the conventional steady-state burning law. Very little can be said about the value of m except that m should be of the order of unity or smaller than unity, but not much bigger than unity. The value of m of a given propellant will probably depend on the operating chamber pressure, and the initial temperature of the solid propellant.

Formulation and Method of Solution

The flow of the burned gas in the uniform cylindrical duct is assumed laminar and axially symmetric in steady-state operation. The present investigation will further be restricted to the case where the square of the Mach number of the gas flow is negligibly small compared to unity. Thus, the steady-state chamber pressure and density of the burned gas are practically uniform. If, in addition, the shift of the equilibrium composition of the combustion gases due to the small pressure variation in the chamber is neglected, the flow of the burned gas is isentropic.

The stability of small periodic disturbances superposed on steady-state flow will be studied. Let the reference pressure, density, and gas velocity be the pressure, the density, and the speed of sound in stagnant burned gas, and write the dimensionless instantaneous values of flow properties as

$$\begin{cases} p = 1 + \Re[\varphi e^{i\alpha t}] \\ \rho = 1 + \Re[\delta e^{i\alpha t}] \\ u = \bar{u} + \Re[\mu e^{i\alpha t}] \\ v = \bar{v} + \Re[\nu e^{i\alpha t}] \\ w = \Re[\chi e^{i\alpha t}] \end{cases} \qquad [6]$$

where u, v, and w are the velocity components in the axial, the radial, and the circumferential directions. $\alpha = \omega + ik$ is the complex quantity whose real part represents the angular frequency ω based upon the reference time which is defined as the time required for the sound wave in stagnant gas to travel a distance equal to the radius of the cylindrical cavity. The imaginary part κ is the damping coefficient based on the same characteristic time. The periodic disturbance is stable, neutral, or unstable according as $k \gtreqless 0$.

The linearized small perturbation equations for the five unknown time-independent functions φ, δ, μ, ν, and χ for the flow system in terms of cylindrical coordinates x, r, and θ are

$$\begin{cases} \left[i\alpha + \bar{u}_x + \frac{1}{r}\frac{\partial}{\partial r}(r\bar{v})\right]\delta + \bar{u}\delta_x + \bar{v}\delta_r + \frac{1}{r}\frac{\partial}{\partial r}(r\nu) + \mu_x + \frac{1}{r}\chi_\theta = 0 \\ (i\alpha + \bar{u}_x)\mu + \bar{u}\mu_x + \bar{v}\mu_r + \bar{u}_r\nu = -\frac{1}{\gamma}\varphi_x - (\bar{u}\bar{u}_x + \bar{v}\bar{u}_r)\delta \\ (i\alpha + \bar{v}_r)\nu + \bar{u}\nu_x + \bar{v}\nu_r + \bar{v}_x\mu = -\frac{1}{\gamma}\varphi_r - (\bar{u}\bar{v}_x + \bar{v}\bar{v}_r)\delta \\ \left(i\alpha + \frac{1}{r}\bar{v}\right)\chi + \bar{u}\chi_x + \bar{v}\chi_r = -\frac{1}{\gamma}\frac{1}{r}\varphi_\theta \\ \varphi = \gamma\delta \end{cases} \qquad [7]$$

where the reference length is taken to be the radius of the cylindrical cavity, and λ is the ratio of the length of the propellant grain to the reference length.

Series solutions in terms of the small parameter $\bar{u}_e$, the axial exit velocity are tried. Hence, let

$$\delta = \delta^{(0)} + \bar{u}_e\delta^{(1)} + \dots$$
$$\mu = \mu^{(0)} + \bar{u}_e\mu^{(1)} + \dots$$

The 0th order solutions are exactly acoustical solutions and the first-order corrections $\delta^{(1)}$, $\mu^{(1)}$, etc., are given in appendix 1. The acoustical solution satisfying the condition of periodicity in circumferential direction, and the condition of vanishing velocity disturbances at $x = 0$ is given as

$$\delta^{(0)} = e^{i\beta\theta} \cdot \cos(\xi x) \cdot [J_\beta(\eta r) + cY_\beta(\eta r)] \qquad [8]$$

where

- β = integers = characteristic constants in θ coordinate
- α^2 = $\xi^2 + \eta^2$ with ξ and η the characteristic constants in axial and radial direction, respectively
- J_β, Y_β = Bessel's function of the first and the second kind of order β
- c = integration constant which is in general complex

and the components of the velocity perturbation are

$$\mu^{(0)} = -\frac{1}{i\alpha}\delta_x^{(0)}$$
$$\nu^{(0)} = -\frac{1}{i\alpha}\delta_r^{(0)}$$
$$\chi^{(0)} = -\frac{1}{i\alpha}\delta_\theta^{(0)} \cdot \frac{1}{r} \qquad [9]$$

As shown in appendix 1, the evaluation of the first-order corrections can be carried out for a given steady-state velocity distribution of a particular rocket system. It is rather fortunate that the magnitude of $\delta^{(1)}$ is expected to be of the order of $|\xi|$ or $|\bar{v}|/\bar{u}_e$ which are small quantities of the order of $1/\lambda$ and are comparable to $\bar{u}_e$ for the practical cases considered here. The first correction term $\bar{u}_e\delta^{(1)}$ may be neglected even without serious error in quantitative results under the assumption of $\bar{u}_e^2 \ll 1$. Therefore, the qualitative stability behavior of the high-frequency small disturbances in systems

with different geometrical configurations will be studied on the basis of 0th order solution, or the acoustical solution.

The acoustical solution as given by Equation [8] should further satisfy two boundary conditions in the radial direction and one boundary condition in the axial direction. The boundary condition at a nonburning cylindrical solid wall is the vanishing of the velocity disturbance in the radial direction, that is

$$\nu = \delta_r = 0 \qquad [10]$$

The boundary condition at a burning cylindrical solid surface is that the fractional increase of the mass outflow from the surface is equal to the fractional increase of the instantaneous rate of burned gas generation. It is assumed that the thickness of the reaction zone is negligibly small compared to the reference length, and the boundary condition will be applied right on the solid surface. Thus

$$\frac{\dot{m}_b(t)}{\bar{\dot{m}}_b} - 1 = \frac{\delta\bar{v}_1 + \nu}{\bar{v}_1}$$

Substitute $\dot{m}_b(t)/\bar{\dot{m}}_b$ from Equation [5] and replace the instantaneous gas pressure by the sum of the mean gas pressure and the complex pressure perturbation; then divide through with δ. The boundary condition on the burning surface is obtained as the following value of the ratio of the perturbations of the normal velocity and the gas density at the burning surface.

$$\frac{\nu}{\delta} = -[1 - \gamma m + \gamma(m - n)e^{-i\alpha\bar{\tau}}]\bar{v}_1 = -B\bar{v}_1 \qquad [11]$$

The gas flow at the axial exit $x = \lambda$ is essentially a subsonic jet subjected to the downstream restriction of a sonic throat in the de Laval nozzle. The oscillations in the radial and the circumferential directions further complicate the situation. It seems that the primary effect of the transversal oscillations in the jet is the modification of the jet boundary, not the mean axial-flow conditions, especially when the time-wise variations of the flow properties at a given position due to the transversal oscillations are much faster than those due to the axial oscillations, i.e., $|\eta| \gg |\xi|$. The axial boundary condition at $x = \lambda$ will hence be given approximately by a one-dimensional analysis of the gas flow in the subsonic jet entering the de Laval nozzle. The major effect of the presence of stagnant gas region, around the rocket wall beyond the end of the propellant grain on the one-dimensional flow, is the modification of the steady-state axial velocity distribution from the jet exit to the nozzle throat. In other words, the effect of the stagnant gas region next to the combustion chamber wall and the nozzle wall is to modify the effective shape of the nozzle (Fig. 1). It is shown in (5) that the one-dimensional analysis of the axial oscillations in the nozzle with the downstream restriction of a nonsingular sonic throat leads to a definite relation between the angular frequency of the axial oscillation and the ratio of the fractional variation of velocity to that of density at the nozzle entrance. This relation for the case of linear steady-state velocity throughout the subsonic portion of the nozzle is determined and presented in graphical form in (5) and is reproduced as Fig. 2. It seems that this relation will not be qualitatively changed when the steady-state velocity distribution in the converging section of the nozzle is not quite linear. Call the ratio $\left(\frac{\mu}{\bar{u}}\right)\Big/\left(\frac{\delta}{\bar{\rho}}\right)$ as $A(\Omega, \bar{u}_e)$, where Ω is the reduced frequency parameter equal to the angular frequency of the oscillation divided by the steady-state velocity gradient in the subsonic portion of the effective nozzle, i.e.,

$$\Omega = \frac{\lambda\xi_r \cdot e}{\left(\frac{2}{\gamma + 1}\right)^{1/2} - \bar{u}_e}$$

Then the axial boundary condition at $x = \lambda$ is

$$\mu/\delta = A(\Omega, \bar{u}_e) \cdot \bar{u}_e \qquad [12]$$

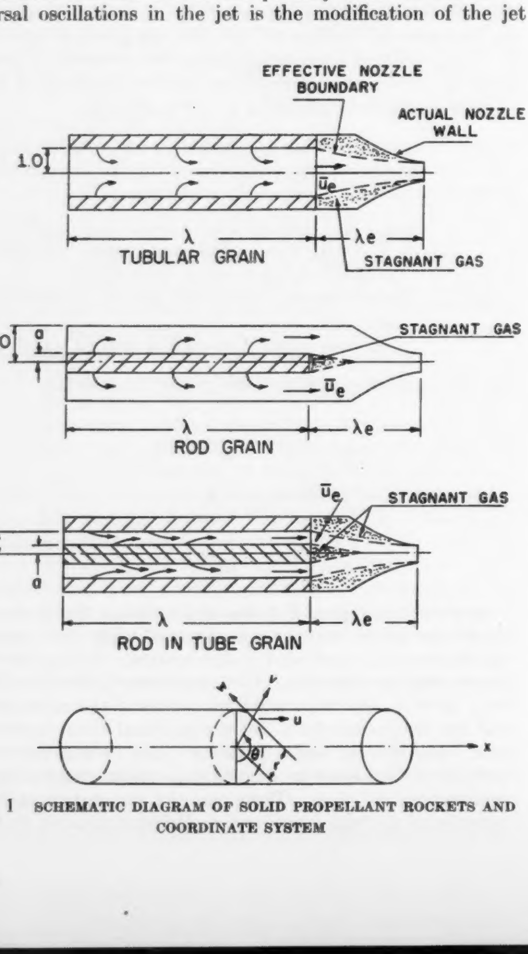

FIG. 1 SCHEMATIC DIAGRAM OF SOLID PROPELLANT ROCKETS AND COORDINATE SYSTEM

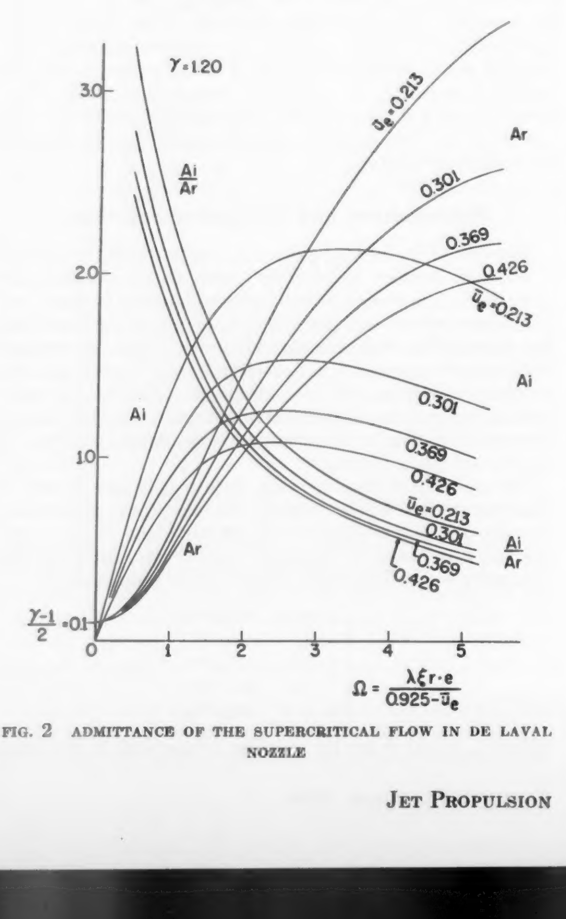

FIG. 2 ADMITTANCE OF THE SUPERCRITICAL FLOW IN DE LAVAL NOZZLE

These three boundary conditions given by Equations [10] to [12] and the relation

$$\alpha^2 = \xi^2 + \eta^2 \quad \text{[13]}$$

provide four relations for the determination of the four complex constants c, ξ, η, and α of a given system. To determine the stability boundary of small oscillations as a relation between m, n, and $\bar{\tau}$ we can put α real and equal to the angular frequency ω.

If the 0th order approximation, i.e., the acoustical solution, is used in the stability boundary calculation, one obtains from Equation [10] the boundary condition at nonburning solid surface with radius r_1

$$J_\beta'(\eta r_1) + cY_\beta'(\eta r_1) = 0 \quad \text{[14]}$$

from Equation [11], the boundary condition at burning surface with radius r_2

$$\eta\,\frac{J_\beta'(\eta r_2) + cY_\beta'(\eta r_2)}{J_\beta(\eta r_2) + cY_\beta(\eta r_2)} = i\omega B\bar{v}_1 \quad \text{[15]}$$

and from Equation [12], the boundary condition at axial exit $x = \lambda$

$$\xi \tan \xi\lambda = i\omega A \cdot \bar{u}_e \quad \text{[16]}$$

where

$$\xi = \xi_r + i\xi_i \qquad \text{and} \qquad A = A(\lambda\xi_r)$$

For the fundamental mode of high-frequency oscillations, ω is of the order of unity, and Equation [16] shows that $\lambda\xi \tan \lambda\xi$ is of the order of $\lambda\bar{u}_e$. The quantity λ is closely related to the ratio of the area of the burning surface to the area of the exit port and is of the order of 10 to 50 for ordinary configurations of solid rockets. Therefore, $\lambda\bar{u}_e$ is of the order of unity and $|\xi^2|$ is of the order of $1/\lambda^2$ which is much less than unity. Equation [13] can thus be written as

$$\eta = \omega\left(1 - \frac{\xi^2}{2\,\omega^2}\right) \quad \text{[17]}$$

and the neutral frequency ω is essentially equal to the real part of η.

If one of the radial boundary conditions is given by Equation [14] over a nonburning surface and the other radial boundary condition is given by Equation [15] over a burning surface, the constant c can be eliminated. The eliminant is

$$\eta\,\frac{J_\beta'(\eta r_2)/J_\beta'(\eta r_1) - Y_\beta'(\eta r_2)/Y_\beta'(\eta r_1)}{J_\beta(\eta r_2)/J_\beta'(\eta r_1) - Y_\beta(\eta r_2)/Y_\beta'(\eta r_1)} = i\omega B\bar{v}_1 \quad \text{[18]}$$

where $J_\beta'(\eta r_1)$ and $Y_\beta'(\eta r_1) \neq 0$.

The right-hand side of Equation [18] is of the order of $|\bar{v}_1|$ which is very small compared to unity. The value of η satisfying Equation [18] is, hence, closely approximated by the real quantity η_0 that makes the left-hand side of Equation [18] vanish. The deviation of η from η_0 is a complex quantity whose magnitude is of the order of $|\bar{v}_1|$. By comparing $\eta - \eta_0$ with Equation [17] we see that $\omega - \eta_0$ is a small quantity either of the order of $1/\lambda^2$ or of the order of $|\bar{v}_1|$. Since η_0 is determined by the radial dimensions only, the neutral frequency ω is also essentially determined by the radial dimensions only.

By expanding

$$\frac{J_\beta'(\eta r_2)/J_\beta'(\eta r_1) - Y_\beta'(\eta r_2)/Y_\beta'(\eta r_1)}{J_\beta(\eta r_2)/J_\beta'(\eta r_1) - Y_\beta(\eta r_2)/Y_\beta'(\eta r_1)}$$

into Taylor series about the point $\eta = \eta_0$, where $J_\beta'(\eta_0 r_2)/J_\beta'(\eta_0 r_1) = Y_\beta'(\eta_0 r_2)/Y_\beta'(\eta_0 r_1)$, and taking only the first term in the expansion, Equation [18] can be rewritten as

$$-\eta(\eta - \eta_0)\cdot\kappa = i\omega\bar{v}_1[1 - \gamma m + \gamma(m - n)e^{-i\omega\bar{\tau}}] \quad \text{[19]}$$

where[4]

$$\kappa = r_2\left\{\left(1 - \frac{\beta^2}{r_2^2\eta_0^2}\right) - \left(1 - \frac{\beta^2}{r_1^2\eta_0^2}\right)\left[\frac{Y_\beta'(\eta_0 r_2)}{Y_\beta'(\eta_0 r_1)}\right]^2\right\} \quad \text{[20]}$$

Substituting η from Equation [17] into Equation [19] and separating real and imaginary parts, one obtains

$$\begin{cases} \gamma(m - n)\sin\omega\bar{\tau} = \\ \qquad -\dfrac{\omega - \eta_0}{\bar{v}_1}\kappa + \dfrac{\Re(\xi^2)}{2\,\omega\bar{v}_1}\cdot\dfrac{2\,\omega - \eta_0}{\omega}\kappa = F \\ \gamma(m - n)\cos\omega\bar{\tau} = \\ \qquad \gamma m - \left[1 - \dfrac{2\,\omega - \eta_0}{\omega}\cdot\dfrac{(\xi^2)}{2\,\omega\bar{v}_1}\cdot\kappa\right] = \gamma m - G \end{cases} \quad \text{[21]}$$

the values of $S = m - (n/2)$ and $\bar{\tau}$ for neutral oscillations of frequency are given by

$$\begin{cases} S = \dfrac{1}{2\gamma}\left[G + \dfrac{F^2}{G - \gamma n}\right] & \text{[22]} \\ \bar{\tau} = \dfrac{1}{\omega}\sin^{-1}\dfrac{F}{\gamma\left(s - \dfrac{n}{2}\right)} & \text{[23]} \end{cases}$$

where the value of $\sin^{-1}$ should be selected in proper quadrant consistent with the sign of $\cos\omega\bar{\tau}$. Let the smallest positive value of $\bar{\tau}$ found from Equation [23] for given value of ω be $\bar{\tau}_0$, then the successive critical values of $\bar{\tau}$ are given by

$$\bar{\tau} = \frac{2h\pi}{\omega} + \bar{\tau}_0 \quad \text{[24]}$$

with $h = 0, 1, 2 \ldots$

Since $m - n$ is of the order of unity, we see from Equation [21] that $\left|\dfrac{\omega - \eta_0}{\bar{v}_1}\right|$ must be of the order of unity, i.e., the angular frequency ω must be very close to η_0. The deviation is of the order of $|v_1|$ which is very small. While the neutral frequency ω varies in the neighborhood of η_0, $(2\,\omega - \eta_0)/\omega$ is practically unity, and it will be clear later that $\left|\dfrac{\xi^2}{2\,\omega\bar{v}_1}\right|$ is a slowly varying function of ω and of the order of unity. Thus G is a slowly varying function of ω since κ is independent of the variation of ω. The minimum value of $S = m - (n/2)$ for all possible frequencies of neutral oscillation of the particular system is, therefore, given practically by

$$S_{\min} = \frac{G}{2\,\gamma} = \frac{1}{2\,\gamma}\left[1 + \frac{\mathscr{I}(\xi^2)}{-2\,\eta_0\bar{v}_1}\cdot\kappa\right] \quad \text{[25]}$$

where the corresponding neutral frequency ω_0 has been replaced by η_0 without serious error. The value of ω_0 is defined by

$$F(\omega_0) = -\frac{\omega_0 - \eta_0}{\bar{v}_1} + \frac{2\,\omega_0 - \eta_0}{\omega_0}\cdot\frac{\Re(\xi^2)}{2\,\omega_0\bar{v}_1} = 0$$

If the value of the over-all pressure index S of interaction of a given propellant is lower than $S_{\min}$ of a solid rocket at a given instant, there cannot be any real solution for the frequencies ω of neutral oscillations. In other words, small oscillations are stable under that configuration. If S of the propellant is bigger than $S_{\min}$, there will, in general, be two real solutions for ω. Call the two solutions ω_1 and ω_2, with $\omega_1 > \omega_0$ and $\omega_2 < \omega_0$. Corresponding to each of the two real solutions for ω, there will be a series of values of the critical time lag $\bar{\tau}$. For example, if the S of the propellant used in a rocket with tubular grain where $\bar{v}_1 < 0$ is slightly bigger

[4] For the reduction of Bessel's function see, for example, Ref. (6).

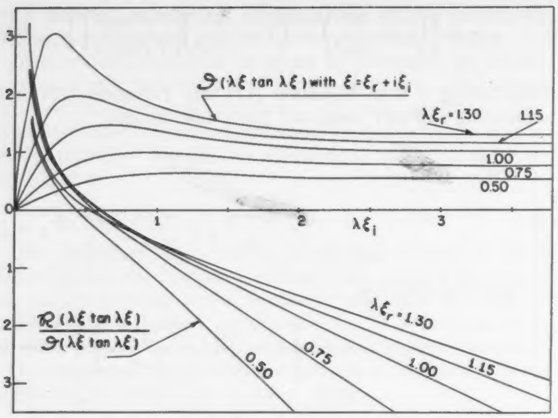

FIG. 3 $\lambda\xi \tan(\lambda\xi)$ AS A FUNCTION OF $\lambda\xi = \lambda\xi_r + i\lambda\xi_i$

than $S_{\min}$ of the rocket at a certain instant, then $\omega_1\bar{\tau}_{01}$ lies in second quadrant and $\omega_2\bar{\tau}_{02}$ lies in third quadrant. Both $\omega_1\bar{\tau}_{01}$ and $\omega_2\bar{\tau}_{02}$ approach π when S approaches $S_{\min}$. If the time lag $\bar{\tau}$ of the propellant in dimensionless form lies in any of the unstable ranges

$$\frac{2h\pi}{\omega_1} + \bar{\tau}_{01} < \bar{\tau} < \frac{2h\pi}{\omega_2} + \bar{\tau}_{02}$$

then oscillations of frequency ω with $\omega_2 < \omega < \omega_1$ are unstable. It should be noted that the unstable range of the dimensional time lag $\bar{\tau}^*$ is

$$\left(\frac{2h\pi}{\omega_1} + \bar{\tau}_{01}\right)\frac{r_2^*}{c_0^*} < \bar{\tau}^* < \left(\frac{2h\pi}{\omega_2} + \bar{\tau}_{02}\right)\frac{r_2^*}{c_0^*} \quad \ldots\ldots [26]$$

which increases continuously as r_2^* increases in the course of operation of the solid rocket with tubular grain. It will be shown later that $S_{\min}$ of such a rocket decreases in the course of operation. Therefore, once the value of $S_{\min}$ of a given rocket becomes smaller than the actual value of S of the propellant, the actual time lag $\bar{\tau}^*$ of the propellant will sooner or later get into the unstable range as indicated by Equation [26]. By then, small oscillations become unstable. The relative magnitude of the values of $S_{\min}$ of solid rockets of different geometrical configurations are, consequently, good indications of the relative stability of small oscillations of random frequencies in these systems. Equation [25] will then serve as the basis of the following discussions for the relative stability of solid rockets of different geometrical configurations. From another point of view, the value of $S_{\min}$ as given in Equation [25] represents the largest tolerable index of interaction in order that the stability of small high-frequency disturbances is unconditionally guaranteed in the particular configuration, irrespective of the time lag of the propellant under the particular operating conditions.

(End of Part I)

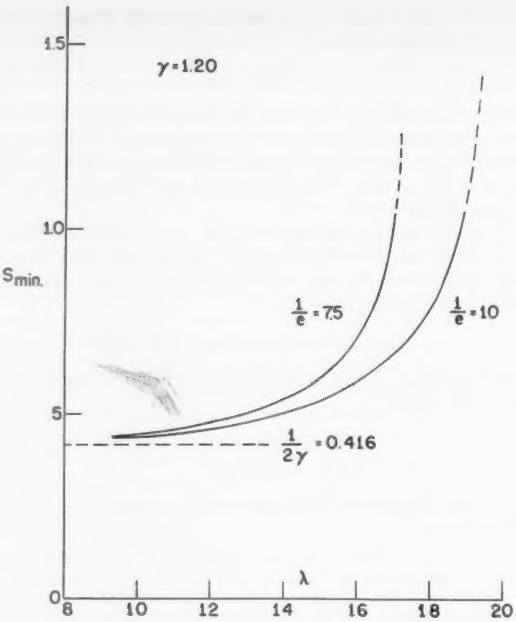

FIG. 4 MINIMUM PRESSURE INDEX OF INTERACTION $S_{\min}$ FOR TUBULAR GRAIN WITH LENGTH TO RADIUS RATIO λ OF THE INTERNAL BURNING SURFACE

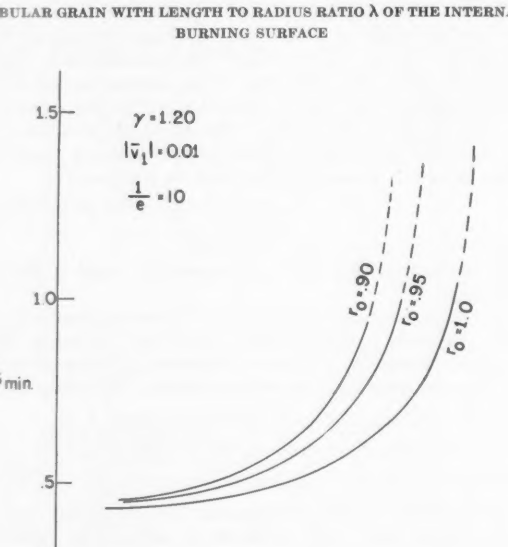

FIG. 5 EFFECT OF END SEAL WITH RADIUS EQUAL TO r_0 TIMES THE PORT RADIUS OF THE TUBULAR GRAIN

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Technical Notes

Terminology and Nature of Atmospheric Shells

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IN THE present terminology, which is being used by the Geophysics Research Directorate (GRD) of the Air Force Cambridge Research Center (AFCRC), the atmospheric shells are distinguished on the basis of changes in temperature gradient with height in the atmosphere as indicated in the diagram. Each shell or layer is called a "———— sphere"; for example, the troposphere. The upper boundary of any layer is given the corresponding name with the suffix "pause"; for example, the tropopause. The boundary surfaces may actually have considerable thickness.

Our ordinary "weather" is confined to the lowest layer, the *troposphere*, which is in convective (turbulent) thermal equilibrium with the sun-warmed surface of the earth, somewhat like a kettle of water on a hot stove. The adopted standard "lapse rate" (negative of temperature gradient) is constant at 65 C/km throughout the troposphere. What "heats the stove" or warms the surface of the earth is solar radiation in the visible region (ordinary sunlight), which is the only portion of the spectrum which readily reaches to the surface through the entire atmosphere without being absorbed along the way to produce direct atmospheric heating. (Even in cloudy weather most of the visible light reaches the ground—as "daylight"—after being scattered in all directions by water droplets in the clouds and by the ever-present air molecules and dust and haze particles in the troposphere.) Various portions of the incoming ultraviolet solar radiation as well as the outgoing infrared black-body radiation from the warm (relative to absolute zero!) surface of the earth are also available to heat the various layers of the atmosphere, as we shall see. As for the outgoing infrared radiation, it is largely absorbed in the two lowest layers by water vapor, carbon dioxide, and ozone to heat these layers. However, it is immediately reradiated in all directions to cool these layers. Averaging over diurnal, synoptic, and seasonal changes, each portion of the atmosphere is in thermal equilibrium losing just as much heat by conduction, convection, and radiation as it gains by these processes but usually in different proportions and at different wave lengths. The tropopause or top of the troposphere is the boundary level at which the lapse rate more or less abruptly becomes approximately zero, usually at about 11 km or 36,000 ft at middle latitudes. Here the temperature is comparable to the coldest recorded arctic winter temperatures and the pressure is about one fourth of the 1013-mb surface average, in accordance with the aviator's rule of thumb of an exponential decrease in pressure with altitude by a factor one half for every 18,000 ft.

The *stratosphere* is the layer above the troposphere in which the standard atmosphere has constant temperature 216.66 K (i.e., zero lapse rate, and stable) equal to that of the tropopause up to about 32 km (105,000 ft) at middle latitudes. Actually, the constancy of temperature in the stretosphere is an idealization at best, since individual observed temperature curves often show wide deviations, particularly at high and low latitudes. The stratosphere is thick over the poles and thin or even nonexistent over the equator owing to the marked rise in elevation of its base, the tropopause, relative to its top, the stratopause, in going from poles to equator. The temperature is held roughly constant in the stratosphere due to moderate warming by the residual absorption of near ultraviolet (2000–3000 A) solar radiation by ozone (O_3), which has its maximum percentage molecular concentration in the upper stratosphere at about 25 km (82,000 ft). This latter height was slightly exceeded in the recent record-breaking manned flight in the Douglas Skyrocket. The lowest portion of the stratosphere (also the topmost troposphere) is characterized by high winds including the jet stream, by clear air turbulence, and by the highest cirrus clouds. In its upper regions are located the rare nacreous clouds and the ends of the trains of those meteors that are bright enough to be visible in daylight.

The *chemosphere* is a relatively hot layer lying between two cold regions having about the same temperatures, the upper one being the chemopause at about 80 km and the lower one the stratosphere. The heating is due to absorption of sunlight by the tenuous upper fringe of the ozone layer which "skims the cream" of the near ultraviolet energy before it can reach the main concentration of ozone which lies below in the stratosphere. At the temperature maximum, which occurs near the middle of the chemosphere at about 50 km, the atmosphere is comparable with desert air at sea level: say 20 F to 100 F, clear and dry, but a thousand times less dense. Below the temperature maximum there is a stable region with a temperature gradient of from +2 to +4 C/km. Above, there is a drop in temperature to the minimum, as mentioned, the lapse rate being less than half as large as that in the troposphere. (In Fig. 1 it appears to be larger owing to the compression of the vertical scale with height as the cube root of the height.) According to this latter lapse rate the upper portion of the chemosphere should be a region of turbulence relative to the regions of positive temperature gradient immediately above and below. Observations of twisted meteor trails and undulating noctilucent clouds indicate that it is actually such a region. Recent rocket air sampling has shown that, despite this turbulent mixing, there is some diffusive separation as low as 70 km where helium was found from one rocket flight to have doubled its normal molecular concentration relative to the much heavier molecular nitrogen. (The thoroughly mixed lower atmosphere may be likened to homogenized milk; diffusive separation would correspond to some of the less dense cream floating to the top.) The chemosphere is so named for the large number of chemical (mostly photochemical) processes that are known or presumed to be occurring within it. The upper 10 or 15 km of the chemosphere is the region of disappearance of the ordinary meteors and of the recently discovered, enormously intense, infrared night air-glow radiation of the hydroxyl radical, OH; also of the daytime D-layer of ionization (see Fig. 1) that tends to absorb radio waves and which produces radio blackouts at times of bright flares in the solar chromosphere.

The *ionosphere* extends from the temperature minimum near 80 km to a possible temperature maximum of several thousand degrees at around 400 km. The ionosphere is known better than its great height and thickness would suggest, owing to

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Editor's Note: This section of the Journal is open to short manuscripts describing new developments or offering comments on papers previously published. Such manuscripts are published without editorial review, usually within two months of the date of receipt. Requirements as to style are the same as for regular contributions (see first page of this issue).

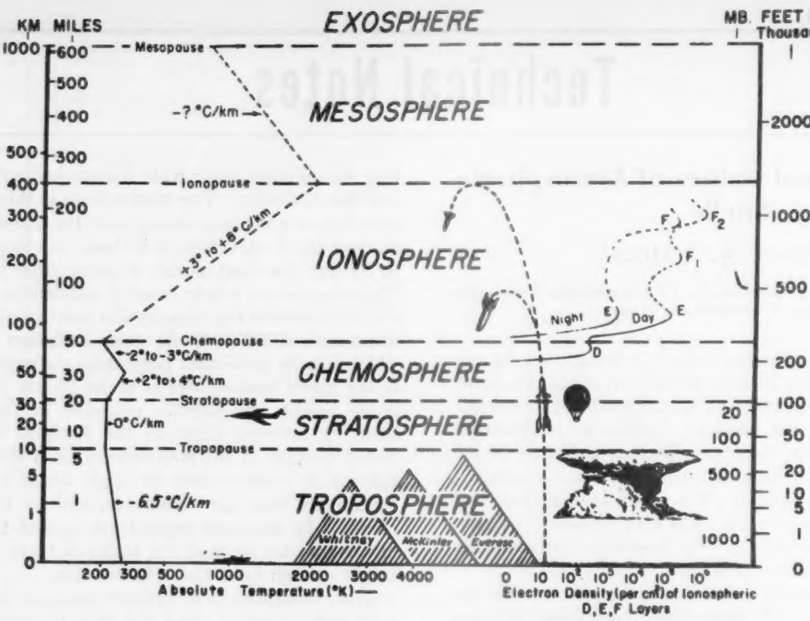

FIG. 1 SCHEMATIC DIAGRAM OF THE GRD SYSTEM OF ATMOSPHERIC NOMENCLATURE

the long use of the powerful method of ionospheric probing by high-frequency radio waves. Also, meteors, aurorae, and rockets regularly give us information about the lower portion; and the record-breaking flight made by a two-step rocket consisting of a WAC Corporal fired from a V-2 rocket in the chemosphere actually reached the top of the ionosphere (see Fig. 1). The name "ionosphere" was given by Watson Watt in the early days of radio because of its layers of ionization that reflect all but the shortest of high-frequency radio waves, and thereby make long distance short-wave radio communication possible. The daytime E, F_1 and F_2 layers and the nighttime E and F ionospheric layers, which range in height from 100 to 300 km, are indicated in Fig. 1, together with their average electron concentrations. The E-region of the lower ionosphere is the level of the upper or beginning ends of typical meteor trails, of the bottoms of the usual aurorae, and of the red bottoms of bright auroral rays and draperies. The ionosphere as a whole is the domain of the normal aurorae and of many of the night air-glow emissions. Essential changes take place in the character of the atmosphere in the ionosphere. For example, molecular oxygen (O_2) which represents 21 per cent by number of the composition of the atmosphere, is completely dissociated into atomic oxygen (O) in the E-layer. This doubles the number of oxygen particles and correspondingly lowers the average molecular weight in the ionosphere. Also the concentration of charged particles (electrons and ions) becomes important not only in its effect on radio waves, but also on the physics and chemistry of the atmosphere itself and on the latters' response to outbursts of corpuscles and ultraviolet radiation from the sun. Ionization (the stripping off of electrons from atoms and molecules to form ions and free electrons) not only increases the number of gas particles but also makes the air conductive and allows huge sheets of current to flow. With increasing height in the atmosphere, molecules of the air can lose heat more readily to outer space. More than counterbalancing this cooling effect with height in the case of the particular atmospheric composition and density prevailing in the ionosphere are the increased amounts of incoming solar radiation at all wave lengths available for the molecules to absorb. The net result is an average temperature gradient of the order of +2 to +8 C/km leading to a temperature of possibly as much as several thousand degrees at the top of the ionosphere (ionopause), as judged by the excitation required by the observed spectra of the aurorae.

The *mesosphere* is the layer intermediate between the ionosphere and the outermost layer, the exosphere. It is considered to be a region of high but decreasing temperature. The electron density is high, but probably not as high as in the ionosphere. Percentagely, however, the level of ionization may actually be higher owing to the hundredfold drop in atmospheric density within the mesosphere. Far ultraviolet (<2000 A) sunlight is very energetic and is peculiarly effective in producing ionization rather than mere dissociation of molecules as occurs characteristically in the chemosphere due to the less energetic near ultraviolet. Although the far ultraviolet carries the greatest energy, it is also absorbed most readily by overlying gases and scarcely reaches the chemosphere. By contrast, in both the ionosphere and the mesosphere there is so little overlying gas that the far ultraviolet radiation penetrates sufficiently to produce copious ionization. The mesosphere is the region of the highest aurorae, the sunlit aurorae. It is from the spectra of these that we know that ionospheric and mesospheric temperatures must be high.

The *exosphere* is defined as the outermost fringe region of the atmosphere above the "critical level" for escape from the earth's gravitational field. The "mean free path" or average distance a molecule travels between collisions varies from 10^{-5} cm at sea level, 10^{-3} cm at the stratopause (32 km), 0.5 cm at the chemopause (80 km), 0.2 to 0.6 km at the ionopause (400 km), to many kilometers at 1000 km and increases rapidly thereafter. Thus, at some level above the mesosphere a fraction $1/e(\sim 1/3)$ of fast neutral atoms and molecules moving upward will never experience a collision within the atmosphere and thus will escape. This level is defined as the critical level and is believed to occur for a typical particle in the vicinity of 1000 km. Strictly, heavier particles like N_2 which move more slowly will have a higher critical level and light; fast particles like hydrogen and helium will escape at lower levels. (Owing to the earth's magnetic field, it is practically impossible for ions to escape at all.) Theoretically, the atmosphere should gradually merge with interplanetary space. We have so far been able to detect the atmosphere (by the highest sunlit aurorae) to 1200 or 1300 km.

Both the standard atmosphere and atmospheric terminology are in an unsatisfactory state. The "standard atmosphere" referred to is the present ICAO (International Civil Aeronautics Organization) standard approved in 1952, and corresponds to latitude 50°, approximately. It differs just enough to invalidate the tables of the NACA standard

atmosphere of 1925. Since the ICAO standard extends to only 60,000 ft (18.3 km), an upward extension is needed. An unofficial Air Force and U.S. Weather Bureau group is in the process of making this extension on the basis of the best rocket, balloon, searchlight, meteor, anomalous sound, etc., data available. The stratosphere as defined above is expected to lose some 8 or 12 km to the chemosphere since the approximate isothermal condition does not actually extend to 32 km, even in middle latitudes. As to terminology, in response to a paper by H. Flohn and R. Penndorf (*Bull. Amer. Meteor. Soc.*, vol. 31, p. 71, 1950), two systems were devised independently and essentially simultaneously in 1950 by N. C. Gerson of GRD and by S. Chapman of Oxford. The former as presented above is simple and has been widely used, especially in the United States. The latter, which consists of several different systems each depending upon a different physical quantity such as temperature, composition, ionization, etc., has been widely used and has been in considerable part recommended for trial use by the IUGG (International Union of Geodesy and Geophysics). Unfortunately, Chapman's "mesosphere" is identical in meaning with Gerson's "chemosphere" as defined above, Chapman's chemosphere includes the GRD chemosphere but is more inclusive. Thus, these two terms lead to confusion and should be replaced. Also there is no IUGG approved term for the 400–1000 km region, Chapman's "suprasphere" being "viewed with reserve." The same altitude limits 0, 11, 32, 80, 400, and 1000 km are recognized in both systems at present, but both are concerned with changes in physical conditions rather than with arbitrary heights in km.

On the Decompression of a Punctured Pressurized Cabin in Vacuum Flight

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Nomenclature

A = area of hole, ft^2

F = $\left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}}$

g = 32.2 lbm ft/lbf sec^2

k = $\frac{c_P}{c_V}$ = ratio of specific heats

P = pressure in cabin at any time, lbf/ft^2

P_1 = initial pressure in cabin, lbf/ft^2

P_2 = final pressure in cabin, lbf/ft^2

R = specific gas constant = $(1544)\ \frac{\text{lb-lbf}}{\text{mole}°\text{R}} \times \frac{1}{(M.W.)\frac{\text{lbm}}{\text{mole}}} = \frac{1544}{M.W.},\ \frac{\text{ft-lbf}}{°\text{R-lbm}}$

where $M.W.$ is the molecular mass of the gas in question, lbm/mole

T = temperature of gas in cabin at any time, °R

T_1 = initial temperature of gas in cabin, °R

t = time in seconds

t_2 = time elapsed from the moment of puncture till P_2 is attained, sec

V_c = volume of cabin, ft^3

W = mass of gas in cabin at any time, lbm

W_1 = mass of gas in cabin at moment of puncture ($t = 0$), lbm

w = mass of gas through hole, lbm

Introduction

The time interval which elapses before a dangerously low pressure is reached in a rigid pressurized cabin which is suddenly punctured in vacuum space, is a critical quantity in designing the appropriate warning devices and countermeasure systems.

The object of this paper is to derive an equation giving a conservative estimate for this time as a function of initial and final pressure, area-of-hole and volume-of-cabin for any given gas at a fixed initial temperature.

The derived function is plotted for air at $T_1 = 500°$ R.

Assumptions

It will be assumed that the flow is isentropic, that the expansion of the gas in the rigid cabin is isentropic, that the hole can be treated as a nozzle of negligible length, that there is no addition of gas to the cabin, and that the gas involved behaves like a perfect gas. It is also assumed that the flow consists of a sequence of steady states and that the velocity of the gas as it approaches the hole is negligible so that Equation [1] holds at all times.

Since the pressure outside the cabin is essentially zero, the ratio of the ambient pressure over the cabin pressure will always be less than the critical ratio $(2/(k+1))^{k/(k-1)}$ and therefore the maximum mass-rate of flow through the hole will prevail at all times. Also the cross-sectional area of the hole can be thought of as equivalent to the throat area of the negligible length nozzle.

These assumptions will yield the minimum time required for the pressure to reach the lower value P_2.

Basic Equations

Under the above assumptions the maximum mass rate of flow will be given by

$$\frac{dw}{dt} = APg\left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}}\left(\frac{k}{gRT}\right)^{1/2} \qquad [1]$$

For a derivation of this equation see Ref. 1 at end of paper. The mass of gas in the chamber at any time will be

$$W = PV_c/RT \qquad [2]$$

Also this mass at any time is

$$W = W_1 - w \qquad [3]$$

For an isentropic expansion of the gas in the cabin, the cabin pressure and temperature will be related, at all times, by

$$T = T_1\left(\frac{P}{P_1}\right)^{(k-1)/k} \qquad [4]$$

Equation [2] may be written

$$W = \frac{V_c(P)^{1/k}}{RT_1}(P_1)^{(k-1)/k} \qquad [5]$$

Hence
$$\frac{dW}{dt} = \frac{V_cP_1^{(k-1)/k}P^{(1-k)/k}}{kRT_1}\frac{dP}{dt} \qquad [6]$$

But
$$dW = -dw \qquad [7]$$

Then Equation [1] becomes

$$\frac{dw}{dt} = -\frac{dW}{dt} = -\frac{V_c}{kRT_1}P_1^{(k-1)/k}P^{(1-k)/k}\frac{dP}{dt} = \frac{APgF(k)^{1/2}}{(gRT_1)^{1/2}(P/P_1)^{(k-1)/2k}} \qquad [8]$$

where

$$F = \left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}} \qquad [9]$$

Equation [8] may be simplified to

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[1] Ordnance Engineer, Rocket Branch, Weapon Systems Laboratory.

$$-P^{(1-3k)/2k}\,dP = (kgRT_1)^{1/2}\frac{kAF}{V_c}P_1^{(1-k)/2k}\,dt \quad ..[10]$$

Integrating Equation [10] between limits

$$P = P_1 \text{ when } t = 0, \text{ and } P = P_2 \text{ when } t = t_2$$

and, rearranging, one obtains

$$\left(\frac{P_1}{P_2}\right)^{(k-1)/2k} = 1 + \left(\frac{k-1}{2}\right)(kgRT_1)^{1/2}\,FA/V_c\,t_2 \quad ..[11]$$

This relation applies to all perfect gases.

Decompression Interval for Air

For air

$$k = 1.40$$

$$R = 1544\,\frac{\text{ft-lbf}}{\text{mole °R}} \times \frac{1}{29}\,\frac{\text{mole}}{\text{lbm}} = 53.24\,\frac{\text{ft-lbf}}{\text{°R-lbm}}$$

$$F = \left(\frac{2}{k+1}\right)^{\frac{k+1}{2(k-1)}} = 0.5786$$

Letting the initial temperature T_1 be 500° R

$$(kgRT_1)^{1/2} = (1.4 \times 32.2 \times 53.24 \times 500)^{1/2} = 1095 \text{ ft/sec}$$

$$k - 1/2k = 0.1428$$

$$k - 1/2 = 0.2$$

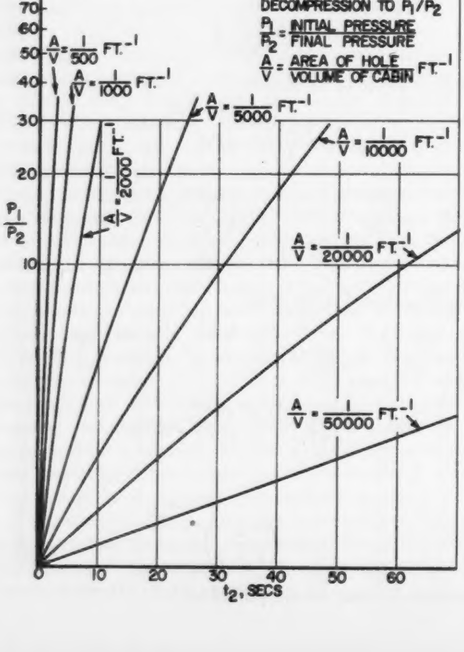
Then Equation [11] becomes

$$\left(\frac{P_1}{P_2}\right)^{0.1428} = 1 + (0.2)(1095)(0.5786)\left(\frac{A}{V_c}\right)t_2 = 1 + 126.7\left(\frac{A}{V_c}\right)t_2 \quad ..[12]$$

A plot of Equation [12] with P_1/P_2 as a function of t_2 for various A/V_c appears in Fig. 1.

Reference

1 "Rocket Propulsion Elements," by G. P. Sutton, John Wiley & Sons, Inc., New York, 1949, p. 53.



Nuclear Reactors for Rockets

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Applied Mathematics Laboratory

Stanford University, Stanford, Calif.

ALMOST as soon as it was realized that energy could be obtained from nuclear reactors, proposals were made to use nuclear energy to heat a working fluid which would propel a rocket. This seemed very attractive since there are many times the energy content of the best chemical fuels in fissionable materials. However, the exhaust velocities which can be obtained depend on the temperature to which the working fluid can be raised. These are limited by the temperatures at which the reactor can be maintained intact and by the temperature difference necessary to insure the heat transfer rates required. In the proposals which the author has seen so far, the limiting temperature has been the melting points (actually the somewhat lower softening temperatures) of the reactor materials. These are not very high for the materials used so far.[2]

This article contains a proposal for getting around these difficulties. The limiting temperature is the boiling point rather than the melting point of the fissionable material.

The rocket motor shown in Fig. 1 spins on the axis AB at as high a velocity as is practicable. The molten fissionable

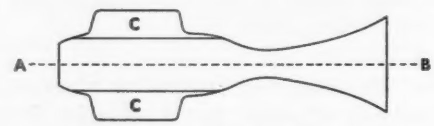

FIG. 1 SPINNING LIQUID REACTOR

material C is held to the wall of the chamber by centrifugal force and operates as a fast neutron reactor. The working fluid, presumably hydrogen, is pumped into the chamber through many small holes in the lining so that it bubbles through the molten fissionable material. It enters cold and reaches its maximum temperature as it bubbles off the surface. Because the working fluid enters cold, the boundary between the fissionable material and the chamber wall is kept at a temperature well below that attained by the hydrogen at the surface of the reactor. Other exposed surfaces in the rocket motor can be protected either by circulating working fluid behind them or by letting working fluid pass through pores in the walls of the chamber.

It is clear that the limiting temperature for this motor is the boiling point of the fissionable material. Uranium has a boiling point of about 2500 C, thorium has a boiling point of over 3000 C, while the boiling point of plutonium has not been published.

A number of different objections to this scheme come to mind.

1 It may be impossible to bubble enough hydrogen through the reactor without blowing it away, and moreover the working fluid might form a volatile compound with the fissionable material.

It is difficult to see how either of these questions can be settled short of experiment, though of course the experiments need not actually involve constructing a rocket motor. It may be pointed out that the centrifugal field can be much

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[1] Acting Assistant Professor of Mathematics.

[2] Uranium melts at 1100 C, thorium at 1845 C; the melting point of plutonium has apparently not been published though presumably it has been determined. If molten fissionable material is to be jacketed by higher melting substances, heat transfer difficulties will arise, and the jacketing materials must not absorb too many neutrons.

stronger than the earth's gravitational field and that this will help prevent blowing away the reactor.

2 The reactor may be impossible to control. The author has seen nothing on the control of fast neutron reactors, but doubts that the control problem will offer fundamental difficulties. One possible method is to make the reactor depend for its criticality on the exchange of neutrons through the wall of the chamber with a small amount of fissionable material kept at a low temperature and controlled by rods.

3 The mechanical difficulties of pumping the hydrogen through the molten material and keeping the motor spinning may be too great. To settle this a detailed design would be required. However, one possibility is to use vanes in the jet to keep the motor spinning, and to use the relative rotation of the motor and the rocket to run the hydrogen pumps.

Kinematics of a Vertical Booster

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An exact solution to the kinematic equation which represents the vertical flight of a booster rocket subject to drag and gravitational force is provided. A relationship between the mass ratio and the burnout velocity is established for the assumption of a constant value of the drag coefficient and the gravitational deceleration.

THE kinematic equation for a booster rocket involves four terms: the forces due to inertia, drag, gravity, and thrust. The usual computational procedure is to solve the equation relating instantaneous velocity to instantaneous mass, neglecting the effect of gravity, and then to add a correction to the final velocity at burnout. The exact solution to the kinematic equation is presented below in dimensionless form. In this analysis it is assumed that the values of the gravitational deceleration and the drag coefficient are constant.

Nomenclature

m = instantaneous mass of booster and payload
m_0 = initial mass of booster and payload
m_e = mass at burnout
t = time
t_B = duration of burning of propellant
F = thrust, assumed constant
c = exhaust velocity, assumed constant
v = booster velocity
K = proportionality constant between drag force and square of booster velocity
g = local value of gravitational deceleration

Subscripts

0 and e refer to initial and burnout conditions, respectively

A force balance on the booster rocket yields the kinematic equation

$$m\frac{dv}{dt} + Kv^2 + mg = F \qquad [1]$$

the mass-time relationship is given by

$$m = m_0 - \frac{F}{c}t \qquad [2]$$

or

$$dt = -\frac{c}{F}\,dm \qquad [3]$$

Now divide [1] by F and then replace dt by [3] to obtain

$$-\frac{m}{c}\frac{dv}{dm} + \frac{K}{F}v^2 + \frac{mg}{F} = 1 \qquad [4]$$

Now substitute

$$\frac{v}{c} = \theta,\quad \frac{c^2K}{F} = \lambda^2,\quad \text{and}\quad \frac{mg}{F} = \rho$$

to obtain

$$-\rho\frac{d\theta}{d\rho} + \lambda^2\theta^2 + \rho = 1 \qquad [5]$$

Now substitute

$$\theta = -\frac{\rho}{s\lambda^2}\frac{ds}{d\rho}$$

to obtain

$$\rho^2\frac{d^2s}{d\rho^2} + \rho\frac{ds}{d\rho} + \lambda^2(\rho - 1)s = 0 \qquad [6]$$

Now substitute $x = 2\lambda\sqrt{\rho}$ to obtain

$$x^2\frac{d^2s}{dx^2} + x\frac{ds}{dx} + (x^2 - 4\lambda^2)s = 0 \qquad [7]$$

The solution to [7] is given by

$$s = AJ_{2\lambda}(x) + B'J_{-2\lambda}(x) \qquad [8]$$

The fractional velocity at burnout θ_e is given by

$$\theta_e = \frac{v_e}{c} = -\frac{x_e}{4\lambda^2}\left[\frac{J_{2\lambda-1} - J_{2\lambda+1} + BJ_{-2\lambda-1} - BJ_{-2\lambda+1}}{J_{2\lambda} + BJ_{-2\lambda}}\right]_{x=x_e} \qquad [9]$$

where

$$B = \frac{B'}{A} = \left[\frac{J_{2\lambda+1} - J_{2\lambda-1}}{J_{-2\lambda-1} - J_{-2\lambda+1}}\right]_{x=x_0} \qquad [10]$$

Finally x_0 and x_e, or ρ_0 and ρ_e, or m_0 amd m_e are related to the duration of burning by

$$x_0^2 - x_e^2 = \frac{4\lambda^2 g t_B}{c} \qquad [11]$$

Thus θ_e and t_B through [9] and [11], respectively, determine x_0 and x_e. For a given value of t_B, a plot of the mass ratio $R = (x_0/x_e)^2$ versus the burnout velocity θ_e may be constructed for varying values of λ.

Received November 10, 1953.

[1] Development Chemist.

Comment on "Methods of Flow Measurement"[1]

F. MINHAS[2]

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IN THEIR paper, "Methods of Flow Measurement," Jerry Grey and F. F. Liu mentioned the principal types of flowmeters in use, with the advantages over each other and the shortcomings in accurate and instantaneous estimation of flow. While keeping in view the authors' discussion of searching for an ideal instrument, I am led to suggest another type of instrument in which an airfoil of finite span be used as a vehicle for the flow measurement. Since the instrument is mechanical, it might be slightly slack in instantaneous flow measurements, depending upon the care in designing.

The flow produces the lift L, the drag D, the moment M, caused by the pressure distribution due to the motion of the

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[1] "Methods of Flow Measurement," by Jerry Grey and F. F. Liu, JOURNAL OF THE AMERICAN ROCKET SOCIETY, vol. 23, May–June 1953, pp. 133–140.

[2] Instructor in Mathematics and Applied Mechanics.

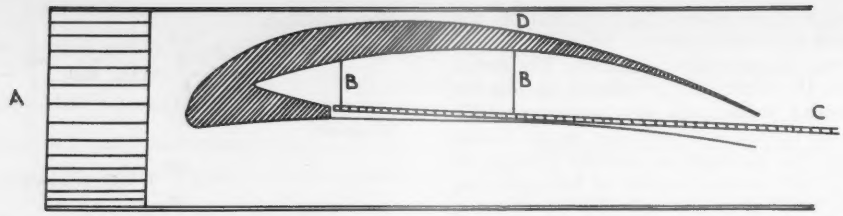

FIG. 1 A: COMB TO MAKE FLOW SMOOTHER. B: STRAIN GAGES. C: SUPPORT. D: AIRFOIL

fluid around the surface of an airfoil, and can be expressed in terms of the lift coefficient C_L, the drag coefficient C_D, and the pitching moment coefficient C_M as follows:

$$L = {}^1/_2\rho^2VSC_L \text{ or mass flow/sec/unit section } = \rho V = \frac{2L}{VSC_L}$$

$$D = {}^1/_2\rho^2VSC_D \text{ or mass flow/sec/unit section } = \rho V = \frac{2D}{VSC_D}$$

$$M = {}^1/_2\rho^2VSC_M \text{ or mass flow/sec/unit section } = \rho V = \frac{2M}{VSLC_M}$$

where S is the fixed area of the finite span airfoil used. The aerodynamic coefficients are functions of the angle of attack, which is to be fixed in this case, Reynolds number *Re*, and Mach number M.

These aerodynamic forces can be determined more readily by the use of wire-strain gages in which specially designed wire is used. It is also necessary for the sake of accuracy to apply a thermocouple to each strain gage station to read the temperatures there, to utilize it for calibration varying with temperature. The principle of measurement of these forces by strain gages is that the airfoil under the influence of these forces must cause bending to the bar on which it is supported, and a pair of strain gages must be struck on opposite sides so that one is compressed when the other is expanded by the flexure of the bar. The direct measurement of the tension in a member cannot give more accuracy (Fig. 1).

A sensitive Wheatstone bridge circuit should be used to detect thermal electromotive forces generated at any junction in the circuit between dissimilar metals. The pairs of lead should be side by side to give thermal compensation, and all joints should be spot welded.

The general principle of the instrument is that the flow to be measured is to be passed over the airfoil of finite span, and the aerodynamic forces in play, namely, L, D, and M, register their magnitudes on the afore-mentioned strain gages, and measure the mass flow/unit area/sec that produced these forces during the calibration of the instrument.

There can be two methods of calibrating the flowmeter in question. (1) Air of known density at known velocity (which on multiplication gives the value of the mass flow/sec through a unit section) to be passed around the airfoil of finite span, and the reading at the strain gage be given the above figure obtained after multiplication. In this way, by varying the velocity of air in the wind tunnel a large range of values can be calibrated as much as desired to be built on the range of the instrument. (2) The other method is to actually measure the aerodynamic force (e.g., lift) which is then doubled, divided by velocity, area of the projection of the airfoil, and coefficient of lift. Thus the figure obtained will represent the mass flow/sec through the unit section at this lift and be figured there. By repeating the above process, the desired range can be covered.

Evidently the efficiency of this instrument depends upon the care with which it is calibrated and the sensitivities of aerodynamic force measuring gages.

The instrument can easily be converted into a continuously recording instrument by the addition of an ink recorder.

Reply by Mr. Grey

Mr. Minhas has suggested an instrument whose features are not unfamiliar to the wind-tunnel aerodynamicist but which is certainly unique when applied to the problem of airflow measurement. There exists, of course, the possibility of using the airfoil in liquid streams, but this application does not appear to be too practical, since not only must its dimensions be kept extremely small, with all the attendant constructional problems, but proper insulation of the strain gages might become troublesome. One drawback is that it measures only local values of mass flow so that, even if zero error in all terms is assumed, over-all mass-flow measurements will be in error.

In view of the comparative complexity of apparatus, calibration, and data reduction procedure, it would appear to this author that the application of a simple instrument such as the pitot-static probe used in conjunction with measurements of stagnation pressure and temperature would provide a far more accurate determination of local mass-flow rates.

INSTRUMENTATION SPECIALIST

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Jet Propulsion News

ALFRED J. ZAEHRINGER, Thiokol Chemical Corporation, *Acting Editor*
JOSEPH C. HOFFMAN, General Electric Corporation, *Contributor*

Rockets and Guided Missiles

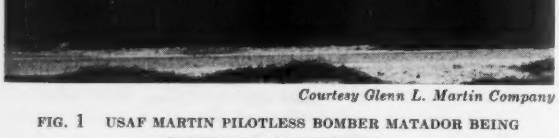
ARMSTRONG-Siddeley of England has a rocket engine which can be throttled by means of regulating the turbo-pump feed unit. Throttling is said to be smooth from zero to full thrust.

ACCORDING to Frank A. Friswold of NACA's Lewis Flight Propulsion Laboratory, rocket engine tests are being successfully monitored by television. To insure the safety of personnel conducting hazardous experiments with propellants, often untried before, a remotely located test cell has been equipped with TV to permit remote viewing of the rocket test. A special enclosure, mounted on skids to facilitate moving, houses the TV camera. It is said that TV can justify itself on a purely economic basis—for example, savings in expensive propellants and new test equipment. NACA is also trying frame-sequential color TV as a possible aid in combustion studies.

SERVICE tests are being conducted on a new antitank missile produced for the U. S. Army Ordnance by the French firm of Sfecmas. The missile is similar to the German X-4 in that the control of the missile is maintained by means of wires which are unwound during the flight of about 2 miles. Stability of the missile is achieved by spin.

PRELIMINARY flight testing of the North American long-range guided missile, Navaho, is to take place shortly at Edwards AFB, Calif., where it has been undergoing static testing.

ROCKET assisted take-off units on the Martin Matador are released without any mechanical linkage or other activating devices according to recent news items. Two simple fittings support each RATO bottle. The RATO thrust holds the bottles in close contact with the fittings during take-off.



Courtesy Glenn L. Martin Company

FIG. 1 USAF MARTIN PILOTLESS BOMBER MATADOR BEING LAUNCHED AT COCOA, FLA.

Fig. 1 shows a RATO launching. When the RATO propellant burns out, the lack of thrust allows the RATO bottles to drop away from the Matador tailcone by air stream pressure. By this time, the prime power plant, an Allison J33-A-37 engine, is in full operation.

THE only guided missile which is equipped with wheels and which can be recovered after a test flight is the Navy's claim for its Chance Vought Regulus. Photographs of recent tests at Muroc AFB show the Regulus being guided in for a landing by a TV-2 control plane. The Regulus uses a parachute to reduce its landing speed and a tri-wheeled landing gear to provide balance and structural protection during landing operations.

ADMIRAL Robert B. Carney, Chief of Naval Operations, revealed that shortly many Navy guided missiles will become operational and all will be phased into fleet activities at about the same time. He emphasized that the concurrent test and application of an entire missile family would result in savings of both tax dollars and time.

THE Defense Department's recent release of information and films on the Nike—besides creating tremendous interest—shed some light on the development of this missile. According to official releases, Bell Telephone Laboratories submitted a proposal on the Nike to the Army Ordnance Corps just five months after first being approached in 1945. Having obtained a development contract, Bell enrolled Douglas Aircraft as a partner in the project, got Douglas to design the missile and launching equipment, and the two companies came up with a finished system five years later. Another indication of the complexities of guidance and control is that the first test firings of the Nike without control took place in 1946, but it was not until two years ago that successful firings of the controlled missile took place. Production is now in the hands of Western Electric Company, Douglas Aircraft, and "several hundred" subcontractors.

FAIRCHILD Guided Missiles Division is developing a guided missile to home on submerged submarines at ranges of 20 miles. The missile is called the "Petrel" and is powered by a Fairchild J-44 turbojet engine.

THE General Electric Company has announced a rocket motor developing 20,000 lb of thrust, easily produced from nonstrategic materials and using conventional propellants. No other details have been given.

CORRECTION: In the September–October 1953 issue of the JOURNAL, page 318, Fig. 1 (Shipboard Launching of Fairchild "Lark") was obtained through the courtesy of the Fairchild Engine and Aviation Corporation.

Jet Propulsion Engines

OFFICIAL USAF announcement has placed the P&W J-57 turbojet engine in the "10,000-lb thrust class." It is reported that with afterburner the thrust can be increased to 15,000 lb. The J-57 is of the split-compressor type and is to power the B-52, F-100, F-101, F-102, and other jet aircraft.

BOEING engineers are testing a new gas turbine engine that weighs only 200 lb but which can power a 55,000-lb vehicle. Developed under Navy sponsorship, the engine measures only 40 in. long, 23 in. wide, and 22 in. high and

EDITOR'S NOTE: The information reported in this Section has been selected from approved news releases originating with the Department of Defense, private manufacturers, universities, etc., and from published news accounts in journals and newspapers. The reports are considered generally reliable, although no attempt has been made to verify them in detail.

delivers 175 hp. Designated as Model 502 by Boeing, the engine is currently being tested in helicopter, airplane, landing craft, and truck-trailer applications. Basically, this engine consists of a gas producer section and a power section. Exhaust gases from the gas producer drive a single axial-flow turbine wheel in the power section which drives an output shaft through a double planetary reduction gear. The engine is started electrically or with compressed air. To keep weight to a minimum, Boeing engineers designed the main parts in aluminum (almost 50 per cent of its total weight).

ROLLS-ROYCE has announced two new models of its Avon turbojet engine: the RA 21 (7150-lb thrust) and the RA 22 (8050-lb thrust).

A NEW centrifugal flow turbojet engine, the Hispano-Suiza Verdon, is now being produced in France. Rated at 7700-lb thrust (without afterburner), the engine boasts a thrust-to-weight ratio of 3.81. A specific fuel consumption of less than 1.0 lb/lb/hr is claimed.

PRATT & Whitney's T-34 turboprop engine is the latest power plant developed for the services that has been released for commercial application both here and abroad. Known as the PT2F-1, the engine will produce 5600 eshp at 11,000 rpm for take-off and has a rated eshp of 4850 at 10,750 rpm. The engine has a 34-in. diameter and is about 158 in. long. Its 2564-lb dry weight gives it a peak performance of well over 2 hp/lb.

THE British firm of Saunders-Roe, Ltd., has announced a pulsejet engine developing 45-lb thrust and weighing only 15 lb. The engine measures $5^1/_2$ in. in diameter and is $47^1/_2$ in. long. The engine was developed to power helicopters.

ANOTHER powerful pulsejet engine, developed by George Schmidt of Brentwood, N. Y., delivers 105-lb thrust and weighs only $17^1/_2$ lb. Stainless-steel tubes mounted in the ejection system are said to provide an "almost silent operation."

A NEW high-power low-weight turbojet engine, the J-46, has been announced by the Navy and Westinghouse Electric Corporation. The new engine will be used to power the Chance Vought F7U-3 Cutlass. The axial flow engine is about $16^1/_2$ ft long and about 3 ft in diameter. The weight is about 2000 lb. Other details have not been released.

Aircraft

REPUBLIC'S newest supersonic all-weather automatic interceptor, the F-103, may be the first conventional aircraft to use ramjet power for tactical operations. The F-103, designed for Mach 3, will be powered by a Wright J-67 turbojet for take-off and landing and a ramjet engine for supersonic operations. A valving arrangement operated by the pilot switches air flow from turbojet to ramjet. Expected weight is about 40,000 lb. Featured is a long, needle fuselage with a delta wing placed well aft.

IT HAS recently been revealed that the Air Force's supersonic, all-weather, jet interceptor, the F-102, has been given its test flight at Edwards AFB, Calif. The delta-wing craft built by Convair is described as an "inhabited missile" because of its near automatic operation. After the craft becomes airborne it is directed to the target by a ground radar control system. In the proximity of the target an automatic guidance system takes over and fires the F-98 Falcon air-to-air rocket missile (built by Hughes) carried by the F-102. After rendezvous the ground control system brings the craft back to home base where the pilot lands the F-102.

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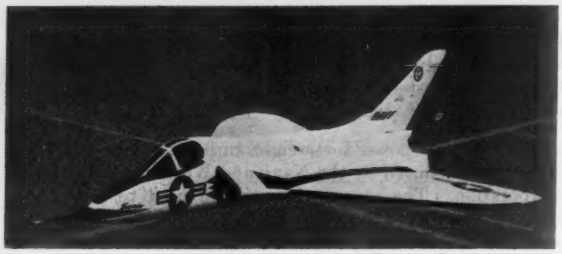

Courtesy Douglas Aircraft

FIG. 2 US NAVY-DOUGLAS F4D SKYRAY SETS NEW WORLD'S SPEED RECORD

A P&W J-57 split-compressor turbojet, delivering about 15,000-lb thrust with afterburner, will power production models of the Douglas XF4D-1 Skyray. The Skyray, shown in Fig. 2, is about 50 ft long with a span of about 34 ft and can fold its modified delta wings for carrier storage. Preproduction models were equipped with a Westinghouse XS40-WE-8 turbojet and afterburner which provided a maximum of about 11,600-lb thrust. A speed record of 753.4 mph for a straight three-kilometer course was set by a Navy pilot, and an average speed record of 728.1 mph for a 100-km closed course was set by a Douglas test pilot. Both flights were made at the Navy Salton Sea test range in California.

AS A part of converting the Air National Guard to jets, the Air Force has started delivering F-86 Sabre Jets to the ANG. Of its 87 tactical squadrons, 60 squadrons are to be all jet-equipped by June 1954.

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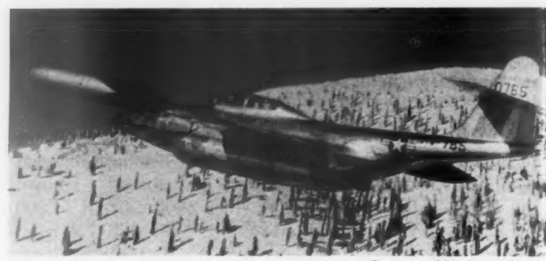
THE NACA Douglas Skyrocket D-558-II set a new unofficial speed record of 1327 mph over the Mojave Desert. The flight was made on October 14, 1953, when the top speed of Mach 2.01 was reached.

NEW models of USAF's Northrop F-89 Scorpion all-weather jet interceptor are being assigned for aerial defense missions from Alaska to Maine. Fig. 3 shows the ship which is in the 600-mph class with a range of about 2000 mi. Power is provided by two Allison J-35 turbojets with afterburners. Service ceiling is about 45,000 ft. With anti-icing systems and elaborate airborne electronic equipment, the F-89 is suited to subzero Arctic climates. The current two-man model carries six 20-mm cannon in its nose section while the F-89D carries 104 2.75-in. The FFAR air-to-air rockets are in wing tip pods.



Courtesy Northrop Aircraft

FIG. 3 USAF-NORTHROP F-89 INTERCEPTOR GUARDS NORTHERN FRONTIER OF THE UNITED STATES

A CESSNA XL-19B, powered by a Boeing XT50-1 turboprop engine set a light-plane altitude record of 37,063 ft. The XL-19B development program is a joint project between the U. S. Army, U. S. Air Force, Cessna, and Boeing.

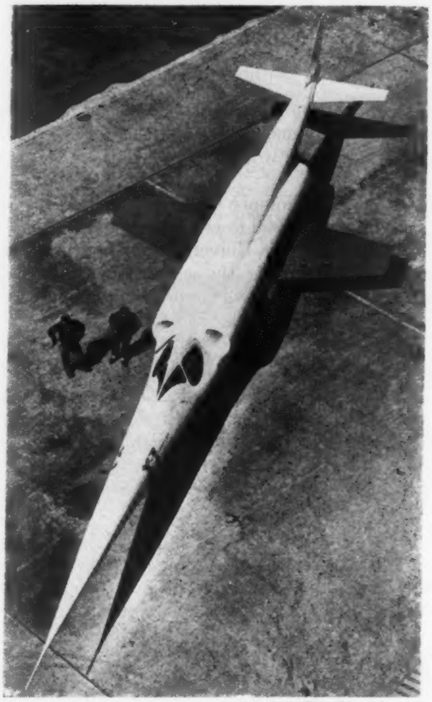
PRODUCTION models of the F-100 Super Sabre are now coming off the assembly lines of North American Aviation. The Super Sabre, Fig. 4, is larger than conventional fighters: 45 ft long, 14 ft high, with a wing span of 36 ft. The service ceiling is above 50,000 ft. Combat radius is more than 500 nautical miles. The wing is thin and swept back. Power is by a P&W J-57 turbojet engine with afterburner. Thrust rating was not announced but the craft is stated to be designed for supersonic speeds in level flight. Considerable weight savings are said to have been made by the extensive use of titanium. The craft is also fitted with many new control systems. At California's Salton Sea, the F-100 recently flew a 15-km course at 754.98 mph with one pass at 767 mph.

Courtesy North American Aviation

FIG. 4 USAF-NORTH AMERICAN SUPER SABRE NOW IN QUANTITY PRODUCTION

ALTHOUGH the mainstay of the Swedish Air Force is the swept-wing Saab-29 jet fighter, it has been disclosed that the Saab-32 "Lancet" with a speed capability of 700 mph is to become the backbone of Sweden's air arm.

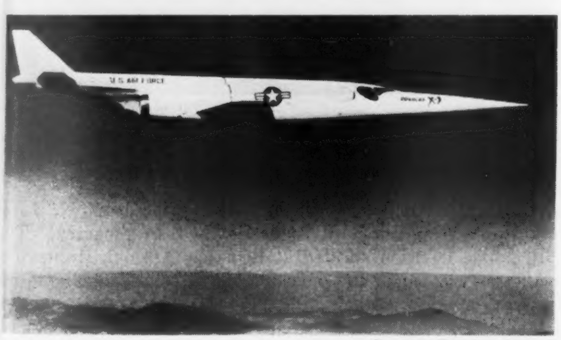
A NEW high-speed aircraft, the X-3 built by Douglas, was revealed by the Air Force. The craft has been under test for over a year and was developed jointly by the AF, NACA, Navy, and Douglas. The craft is being turned over to NACA for research on sustained flight at very high speeds. Figs. 5 and 6 show the striking appearance of the X-3, with a long, slender fuselage and extremely short wings. Length:



Courtesy Douglas Aircraft

FIG. 5 USAF X-3 SUPERSONIC RESEARCH AIRPLANE

66 ft, 9 in.; wing span: 22 ft, 8 in.; tail height: $12^1/_2$ ft from ground; gross weight: about 27,000 lb; payload: 1200 lb plus pilot and fuel; power plant: two Westinghouse J-34-17 turbojet engines plus afterburners. Military restrictions do not permit performance data to be revealed.

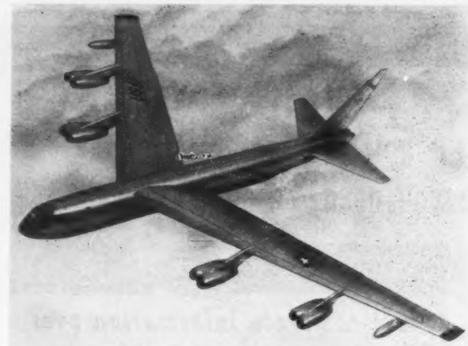


Courtesy Douglas Aircraft

FIG. 6 USAF X-3 BEING FLIGHT-TESTED BY NACA

DOUGLAS has announced that construction of its new Air Force bomber, the B-66B is under way. The bomber, carrying twin jets slung in pods under the wings, places the craft in the "600–700 mph class."

MODEL 707, Boeing's swept-wing jet transport, is expected to be in operation by 1955 according to a recent CAA statement.



Courtesy Boeing Airplane

FIG. 7 NEW USAF-BOEING STRATOFORTRESS B-52A NOW IN PRODUCTION

FIG. 7 shows a model of the Boeing B-52A Stratofortress which is now in production. A new nose and crew compartment is featured with side-by-side seats for pilot and copilot. Eight P&W J-57 turbojet engines are mounted in swept-forward pods located under the swept wing. Auxiliary fuel tanks are located under each wing tip. No details have been released as to speed, range, or other performance characteristics.

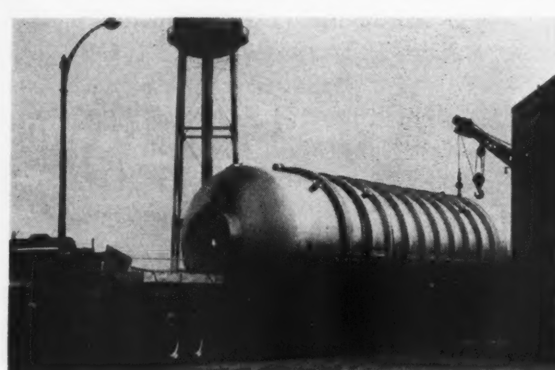
THE Air Force has officially disclosed that the X-1, produced by Bell, hit a top speed of 967 mph in 1948 and in 1949 attained an altitude of 70,140 ft.

New orders and an expanded production program for the Douglas B-66 twin-jet bomber were announced by the Air Force.

Facilities

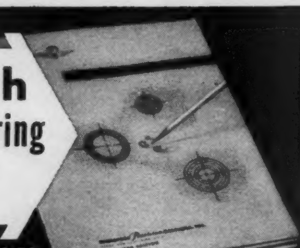
GENERAL Electric will build a new combustion laboratory costing nearly $2.0 million. Located near Schenectady, N. Y., the lab will be completed late in 1954. In addition to office space, work areas include an air compressor and preheater room, combustion test cell, turbine-bucket study area, high-speed test pit, and shock-wave experimental facilities. Areas of study will include jet propulsion, aircraft gas-turbine engines, locomotive and land gas turbines, as well as industrial and domestic applications of combustion equipment.

PRATT & Whitney Aircraft has added a 50-ton altitude chamber (Fig. 8) to its jet testing equipment at the Andrew Willgoos Turbine Laboratory at East Hartford, Conn. The chamber will be used to test full-scale turbojet and ramjet engines and components under various altitude conditions.



Courtesy Pratt & Whitney Aircraft

FIG. 8 ALTITUDE TEST CHAMBER BEING INSTALLED AT P&W TURBINE LABORATORY



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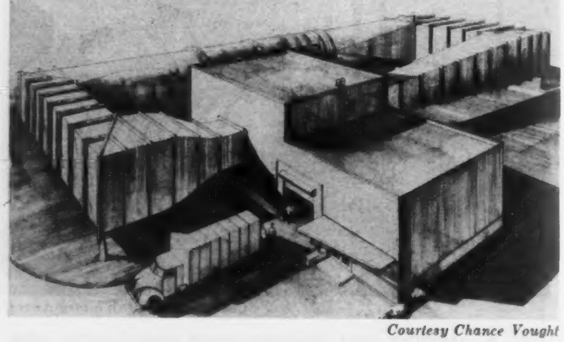
LOCKHEED Aircraft Corporation is the latest firm to enter the nuclear-powered field. According to recent releases, Lockheed has been engaged in a preliminary design study of nuclear-powered aircraft for some time. This work and other nuclear-power programs at Boeing, Convair, Fairchild, General Electric, and Pratt & Whitney are part of the Air Force's broad research activity aimed at eventual development of nuclear-powered aircraft.

A NEW low-speed wind tunnel (Fig. 9), costing about $1.0 million, is to be constructed by Chance Vought Aircraft, Dallas, Tex., for testing aircraft and guided missiles. The speed of the tunnel is about 200 mph and will be used to acquire data in the landing-to-cruising speed range which is now pressing in high-speed aircraft.



Courtesy Chance Vought

FIG. 9 CHANCE VOUGHT TO CONSTRUCT NEW WIND TUNNEL

CONSTRUCTION of $10.5 million Navy plant by Tube Reducing Corporation has begun at Wallington, N. J. The plant will product rocket and RATO bodies in addition to aircraft tubing.

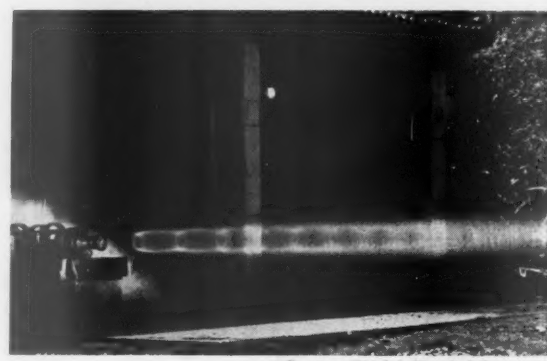
ONE of the first contracts for the specific job of reducing, processing, and analyzing data from tests of rockets and guided missiles has been given to the Land-Air Corporation of Chicago. The firm will operate at the Navy's Guided Missile Test Center at Point Mugu, Calif., through its Data Reduction Office.

THE U. S. Marine Corps has established an unrestricted firing range at 29 Palms, Calif. It was announced that some guided missile training would take place there.

THE Daniel and Florence Guggenheim Foundation is providing a grant of $329,000 to establish an educational and research group at Columbia University for the study of air flight structures, particularly in the supersonic range. Research attention will be given to high-speed aircraft design and to training of men for this field.

THE University of Michigan has dedicated a new building scheduled to handle classified aviation research. The new structure will be operated by the Engineering Research Institute and will study sound transmission characteristics and methods of reducing noise, elimination of ice on aircraft, aircraft armament systems, and evaluation of electronic equipment.

A NEW division to design, develop, and produce guided missiles and pilotless aircraft is being established by Lockheed Aircraft at its Burbank, Calif., plant.



Courtesy Thiokol Chemical Corporation

FIG. 10 STATIC FIRING TEST OF THIOKOL CHEMICAL CORPORATION ROCKET

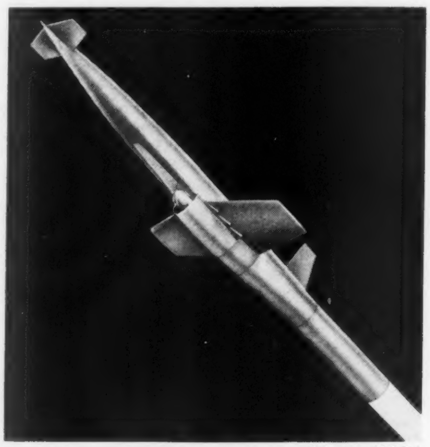
THE Thiokol Chemical Corporation, Trenton, N. J., is now operating three divisions devoted to rocketry. The Redstone Division, located at Redstone Arsenal, Huntsville, Ala., is engaged in the development of solid propellants for guided missiles, rockets, and RATO. The Longhorn Division is reactivating the Longhorn Ordnance Works at Marshall, Tex., and production of solid propellants developed at Redstone is expected shortly. The Elkton Division at Elkton, Md., is concerned with the research, development, and pilot line production of low-cost solid propellants for the Air Force. In addition, propellants for gas generators, auxiliary devices, etc., are being developed. Fig. 10 shows a static test of a large Thiokol RATO unit showing characteristic Mach diamonds.

FORMATION of the Ramo Wooldbridge Corp., Los Angeles, Calif., has been announced. The company, owned

partly by Thompson Products of Cleveland, will do research, development, and manufacturing in guided missiles and other electronic systems.

NACA Leads in High-Speed Flight Research

NACA was founded by Congress in 1915 to "supervise and direct the scientific study of the problems of flight, with a view to their practical solution" and "to direct and conduct research and experiment in aeronautics." An instrumentality of the United States government, it now has an investment of over $200 million in laboratories, buildings, and equipment. Headquarters are in Washington, D. C. Major installations are: Langley Field, Va. (aerodynamics, structures, loads, hydrodynamics); Ames Aeronautical Laboratory, Moffet Field, Calif. (high-speed aerodynamics); Lewis Flight Propulsion Laboratory, Cleveland, Ohio (propulsion);



Courtesy NACA

FIG. 11 NACA RESEARCH MODEL OF RAMJET MISSILE DESIGNED FOR TESTING IN SUPERSONIC TUNNEL AT CLEVELAND LABORATORY OF NACA

High-Speed Flight Research Station, Edwards, Calif. (piloted flight research); Pilotless Aircraft Research Station, Wallops Island, Va. (pilotless aircraft research).

NACA employees now number 7500, nearly one third being professional scientists and engineers. The principal duty of NACA is to do basic and applied research, and the end products are the 800 to 900 technical reports each year—many of them classified.

Research models similar to that shown in Fig. 11 are constructed for tests in a supersonic wind tunnel. The ramjet guided missile shown features a "tail-first" design and has the ramjet engine buried in the fuselage.

Fig. 12 shows a take-off of a research model of a swept-wing, rocket-powered aircraft at Wallops Island. Aerodynamic data are telemetered back to the ground for recording. Such data have been valuable in the design of many of today's high-speed aircraft and missiles.

The NACA has made notable gains in supersonic aircraft research with its X-1, X-2, X-3, and D-558-II Skyrocket. Its achievements in rocket propulsion today will be reflected in the aircraft of tomorrow.

ATLANTIC Research Corporation of Alexandria, Va., has announced expansion of its office and laboratory facilities. Research is carried out in the fields of jet propulsion, high-speed combustion, solid propellant rocketry, shock measurement, and sonar design.

A NEW rocket research center has been announced by the

Courtesy NACA

FIG. 12 ROCKET-POWERED FREE-FLIGHT TEST MODEL LAUNCHED AT NACA RESEARCH STATION

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Air Force. Located at Seymour, Ind., the center will be operated by Standard Oil of Indiana and will specialize in the development of rocket armament and booster rockets for missiles.

Guggenheim Grants Total $36,000 for 1954

Harry F. Guggenheim, president of the Daniel and Florence Guggenheim Foundation, announces that the 1954 Guggenheim Jet Propulsion Fellowships for graduate study in rocket and jet propulsion engineering will total $36,000.

A sum of $18,000 is made available, under the conditions of the grants, to the Daniel and Florence Guggenheim Jet Propulsion Centers at both Princeton University and California Institute of Technology. It is expected that from 18 to 24 grants will be awarded, providing for tuition and living allowances, ranging from $1000 to $2000.

Application blanks for the grants, which go to qualified U. S. residents, have been sent to the major universities and colleges as well as to military establishments and industry engaged in the field. Otherwise they are available from the Foundation, 120 Broadway, New York 5, N. Y. Completed applications must be received by March 1, 1954.

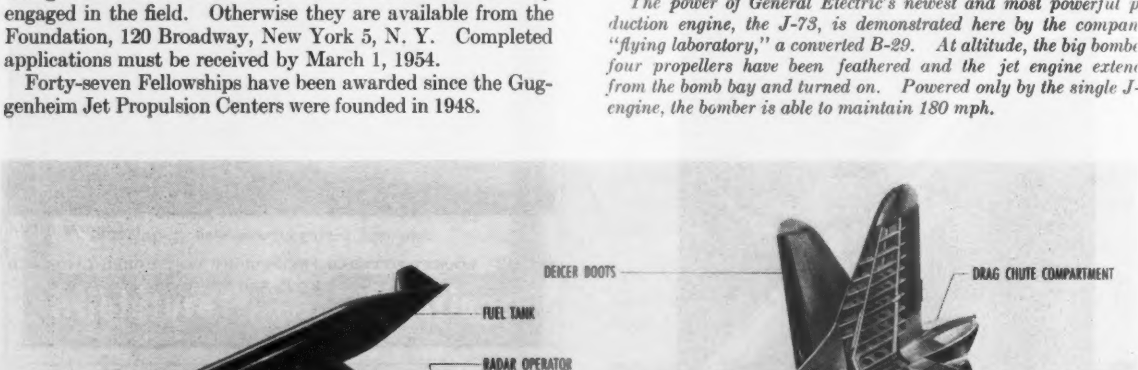
Forty-seven Fellowships have been awarded since the Guggenheim Jet Propulsion Centers were founded in 1948.

NEWS IN PICTURES

Courtesy General Electric

FIG. 13 FLIGHT TEST OF GENERAL ELECTRIC J-73

The power of General Electric's newest and most powerful production engine, the J-73, is demonstrated here by the company's "flying laboratory," a converted B-29. At altitude, the big bomber's four propellers have been feathered and the jet engine extended from the bomb bay and turned on. Powered only by the single J-73 engine, the bomber is able to maintain 180 mph.



Courtesy Lockheed Aircraft

FIG. 14 STARFIRE "OPENS UP"

This first cutaway drawing of the all-rocket-armed Lockheed F-94C Starfire interceptor shows some of its vital organs. Assigned the Air Force mission of defending America against enemy bomber invasion, all-weather Starfires are on 24-hr duty—coast to coast—from the strategic San Francisco Bay region to New York City and the heavy industrial areas of the Northeast and Atlantic Coast. The 600-mph-plus Starfire is armed with four dozen 2.75-in. rockets and packs more than 1200 lb of "brain-like" electronic equipment. Almost entirely automatic, the Starfire can track and shoot down a target its two-man crew may never see, except as a pip on the radarscope. Note the rocket tubes in the nose and in wing pods.

Eighth ARS Annual Convention Draws 1100 to New York

A RECORD seven sessions including 21 technical papers presentations featured the Eighth Annual ARS Convention in New York, Dec. 2 to 4, 1953.

The meeting, again held at the Hotel McAlpin and again in conjunction with ASME's convention, also included the regular Annual Business Meeting, Section Luncheon, and Honors Night Dinner, plus a special trip to the Hayden Planetarium and an all-morning symposium on space flight.

Haley and Porter Elected

Results of the national balloting for 1954 officers announced at the Annual Business Meeting showed that Andrew G. Haley was elected President, with Dr. Richard W. Porter, Vice-President.

Approximately 30 members attended the meeting on the morning of Dec. 2, presided over by President Frederick C. Durant III. Other election results announced were, for three-year terms on the Board of Directors: J. B. Cowen, Director of Government Operations for General Tire & Rubber Co.; George P. Sutton, Supervisor of the Propulsion Section, Aerophysics Lab, North American Aviation, Inc.; and Commander Robert C. Truax, USN.

Fourteen Sections—More Forming

With 61 delegates and guests in attendance, President Durant presided over an informal Section Luncheon on Wednesday, Dec. 2, in the McAlpin Winter Garden.

Represented were eleven of the fourteen Sections now in existence, including the Detroit Section which received its charter during the Convention.

Reports were heard from: *New York*, President Paul F. Winternitz; *Southern California*, Charles E. Bartley; *Cleveland-Akron*, President Darrell C. Romick; *Arizona*, President Anthony R. Tocco; *Indiana*, Elliott Katz; *New Mexico-West Texas*, Secretary Frank L. Koen; *Northeastern New York*, Vice-President Elliot Ring; *Niagara Frontier*, President Willis Sprattling; *Maryland*, Richard C. Lea; *National Capitol*, Andrew G. Haley; and *Detroit*, David Buell.

There were no spokesmen for the Chicago, Alabama, or Northern California Sections. It was announced also that new Sections are about to be established in Waco, Tex., and Parks College, St. Louis, and that interest is being shown in Sections in Cincinnati, Dayton, and the University of Illinois.

Of particular note in the reports of the Southern California, New Mexico-West Texas, Northeastern New York, Arizona, and Cleveland-Akron Sections was the significance and utility of periodic published bulletins. These five sections all publish bulletins, some regularly, some sporadically, but all accented the excellent reaction to them. Mr. Romick pointed out that the bulletin is a fine communications medium for Section business, thereby helping to "streamline" the meetings. Mr. Bartley and Mr. Koen indicated its usefulness as a device for promoting attendance at meetings. Two of the Sections have given their bulletins official names. Northeastern New York has "Impulse" and New Mexico-West Texas has just christened "Missile Away."

Expansion with almost all of the Sections was noted. Arizona, based in Tucson, is up to 74 members and may expand into Phoenix. Indiana has gone from 40 to 100 in one year, Northeastern New York is close to 100, and even newly founded Detroit jumped from 17 to 46 in the six weeks it took to get organized. New York is still tops numerically but is being pushed by Southern California as both Sections have well over 400 members.

Some interesting new ideas were offered during the discussions which took place. Cleveland-Akron has established a committee for the purpose of studying Section activities. This committee may recommend, for instance, the launching of a vocational guidance group for members.

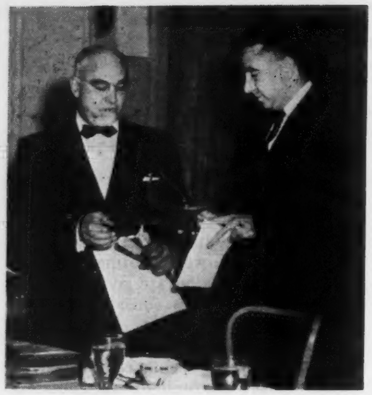
Arizona is soliciting aircraft manufacturers to determine what films are available for meetings and intends to establish a film library bulletin. It was explained later by President Durant that the National Office expects to establish such a list itself and will distribute it to all Sections.

New Mexico-West Texas arranged with the local television station to stage a half-hour discussion program on ARS history and activities. Other ideas from this Section which are worth noting: Mimeographed questionnaires are given to members and the answers considered in arranging future programs; advertising is being solicited for the Section bulletins; a rocket library is being established; a dummy rocket was built to take contributions for postmeeting refreshments.

New Mexico-West Texas also offered criticisms of the JOURNAL. Once brought up, this subject evoked sufficient controversy so that a meeting was called for the following day at which the various views could be aired before Editor-in-Chief Martin Summerfield. The consensus of the criticism seemed to be that a disproportionate amount of space in the JOURNAL was devoted to theoretical papers and that more attention should be given to topics outside of the sphere of pure rocket propulsion. Dr. Summerfield invited the contribution of such papers and indicated a willingness to give them full consideration.

With respect to the two criticism mentioned, Dr. Summerfield pointed out that the editors have never rejected a practical or experimental paper except on objective grounds of technical weakness or absence of originality, and furthermore, that a survey of the six issues of 1953 showed that less than half (16 out of 34) of the papers were confined to rocket power plants. He reminded the Section delegates that it is not the function of the Editor-in-Chief to write the papers, but rather to make available to capable authors the publishing service of the Society. The delegates can help to raise the percentage of practical or experimental papers in the JOURNAL by stimulating their col-

AT DIRECTORS' RECEPTION PRIOR TO HONORS NIGHT DINNER
(*in rear*): Gen. Donald L. Putt; F. C. Durant, III, President ARS; and former President C. W. Chillson. Mrs. Chillson and Mrs. Durant are in foreground



DR. S. S. PENNER RECEIVES THE G. EDWARD PENDRAY AWARD FROM DR. PENDRAY FOR DR. H. S. TSIEN OF THE CALIFORNIA INSTITUTE OF TECHNOLOGY

leagues to write and submit such papers. This he urged them to do.

West Coasters Sweep ARS Awards at Honors Night Dinner

Honors were bestowed on four Californians at the ARS Annual Honors Night Dinner on Thursday evening, December 3, in the Ballroom of the Hotel McAlpin.

The award presentations, made before 306 members and their guests, went to David A. Young of Aerojet-General Corporation, Azusa, Calif.; Charles E. Bartley of Grand Central Aircraft, Glendale; Dr. H. S. Tsien of California Institute of Technology, Pasadena; and Alfred D. Goldenberg of Palm Springs.

Mr. Young, Principal Engineer in charge of special products for Aerojet, received the Robert H. Goddard Memorial Lecture Award from Mrs. Goddard for his contributions to the development of rocket propulsion in the liquid propellant field. He was cited particularly "for the successful development of a large thrust liquid-hydrogen, liquid-oxygen rocket engine, the first anywhere in the world."

C. N. Hickman presented the Hickman award to Mr. Bartley, Director of the Rocket Division at Grand Central Aircraft, for his work in the field of solid propellants. His citation read in part: ". . .for his persistent efforts over many years in pointing out the advantages of solid propellant rocket motors for guided missiles. . ."

Dr. Tsien, recipient of the G. Edward Pendray Award from Dr. Pendray for outstanding contributions to rocket and jet propulsion literature, is the Robert H. Goddard Professor of Jet Propulsion at California Institute of Technology. He has had 44 articles published in technical journals over a period of fifteen years and is responsible for defining the field of "Physical Mechanics" (ARS JOURNAL, January-February 1953, p. 14), which predicts the engineering behavior of matter in bulk from the microscopic properties of its molecular and atomic constituents. Dr. S. S. Penner, an associate of Dr. Tsein, received the award in the latter's absence.

The Annual Student Award was presented by Dr. Richard W. Porter, Chairman of the Awards Committee, to Alfred D. Goldenberg, who founded the first Student Rocket Society in the United States. Mr. Goldenberg's award was presented to the vice-president of the organization, Mr. Sheldon Deretchin of Brooklyn, N. Y.

Five other prominent figures in the jet propulsion field were honored with Fellow Memberships in ARS. They were: Dr. Luigi Crocco of Princeton University; Dr. R. E. Gibson, Applied Physics Laboratory, Johns Hopkins University; Dr. C. C. Furnas, Cornell Aeronautical Laboratory, Buffalo, N. Y.; Mr. J. W. Mullen of Experiment, Inc., Richmond, Va.; and Mr. William M. Smith of Bell Aircraft Company, Buffalo, N. Y.

Putt Points Out Rocket's Military Future

As far as Lt. Gen. Donald L. Putt is concerned, it's not a question of *whether* rocket engines will take over for reciprocating and turbojet units, but *when*.

The Commander of the U. S. Air Force Air Research and Development Command outlined his views before the ARS Honors Night Dinner audience from a dais which included President Durant and President-elect Haley; ARS Directors Pendray and Porter; award recipients Young, Bartley, Penner, and Deretchin; Mrs. Robert H. Goddard; C. N. Hickman; S. Paul Johnston, director of the Institute of the Aeronautical Sciences; Frederick S. Blackall, jr., President of The American Society

DR. C. N. HICKMAN PRESENTS THE HICKMAN AWARD TO C. E. BARTLEY, DIRECTOR OF THE ROCKET DIVISION AT GRAND CENTRAL AIRCRAFT

of Mechanical Engineers; and Dr. Wernher von Braun.

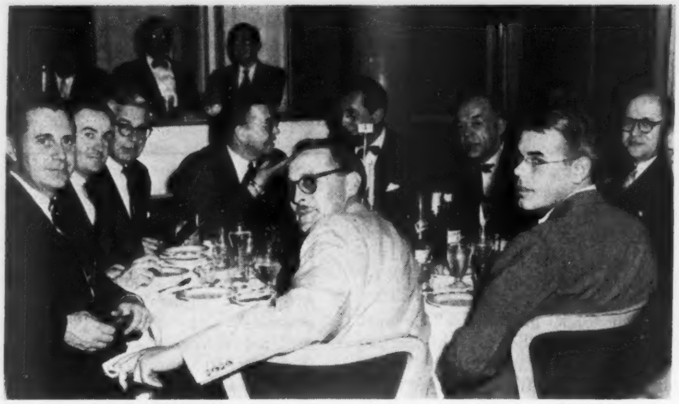
General Putt stated that once fuel-economy problems have been solved, the only limitations to rocket-powered aircraft would be structural and "aeromedical." He later said that "the military airplane, as we know it, will eventually be relegated to a mere logistical vehicle" as the bomber is replaced by the guided missile and guided rockets take over for interceptors and antiaircraft batteries.

Putt pointed to the improvements which have been made since World War II in rocket fuels, materials for power plants, rocket design, and principally in guidance and control. He stated that electronic components were the weakest parts of German guided missiles and that by contrast American engineers have excelled in this aspect of the missiles. Further refinements, he said, such as printed circuits and miniaturization, better gyroscopes and similar components make the advances in guidance probably the biggest step forward in missile development since the end of World War II.

Although a number of our current missiles are equipped with turbojet engines, General Putt said, there is little doubt that the missile of the future will use rocket power exclusively. Considerable work has been carried on by all three Services, and some of the missiles have advanced so far

MRS. ROBERT H. GODDARD PRESENTS THE GODDARD MEMORIAL LECTURE AWARD TO DAVID A. YOUNG. AT LEFT IS GEN. PUTT, MAIN SPEAKER FOR THE EVENING, AND AT REAR IS PRESIDENT DURANT

PRESIDENT DURANT CONFERS A FELLOW MEMBERSHIP UPON WILLIAM M. SMITH OF BELL AIRCRAFT COMPANY



ONE OF THE 30 TABLES AT THE HONORS NIGHT DINNER SHOWING, COUNTER-CLOCKWISE FROM LOVELL LAWRENCE (*center*): R. B. MARSTON, ROCKEFELLER BROS., INC.; C. W. NEWHALL, JR., FLIGHT REFUELING, INC.; M. B. GORDON, ROCKEFELLER BROS., INC.; R. W. YOUNG, RMI; J. BREWSTER, REPUBLIC AVIATION CORP.; R. F. ADAMS, ATTORNEY FOR MATHIESON CHEMICAL CORP.; L. K. HERNDON, MATHIESON CHEMICAL CORP.; AND E. E. FRANCISCO, WSPG

that it has become feasible to establish operational guided missile squadrons.

NOTICE TO READERS: An account of the Space Flight Symposium and a technical summary of the papers presented at the Convention will appear in the next issue of JET PROPULSION.

Section Doings

New York. About 40 members attended a November 20 meeting in the Engineering Societies Building at which the Annual Business Meeting took place. A British Air Ministry film, "Rocket Flight," dealing mostly with German missile developments during World War II, was also shown.

Northeastern New York. Programs for 1954 are scheduled all the way to March 16 when M. J. Zucrow of Purdue University, ARS Director, will deliver a talk on "Jet Propulsion." Paul Hartek of Rensselaer Polytechnic Institute will present "Chemistry of the Upper Atmosphere" on Feb. 16, and C. Mannal of GE's Knolls Atomic Power Lab is scheduled for a Jan. 19 paper on "Application of Nuclear Energy to Rocket and Jet Propulsion." The Section moved over to RPI in Troy for an Oct. 21 meeting at which John B. Duryea of GE's Guided Missile Dept. talked on "Electronic Brains for Rockets and Space Ships" before 125 students and members. Anthony J. Nerod, manager of the Mechanical Investigations Section of GE's Research Laboratory, spoke on "History and Development of Turbojets" on Nov. 17.

Cleveland-Akron. At a recent meeting, Col. Norman C. Appold, Chief of the Powerplant Laboratory at Wright Air Development Center, delivered a talk on "Military Potential of Rocket Engines." Col. Appold had three liquid rockets on display during his lecture, including an early one used by Dr. Robert H. Goddard.

Maryland. The initial meeting, after the Baltimore-Washington Section was split into a Maryland and a National Capitol Section, took place at Munder's Grotto in Baltimore on November 3. It was a dinner meeting attended by approximately 60 members and guests, and was presided over by Ivan Tuhy, the newly elected President. Guest speaker was Andrew G. Haley, who had chartered a bus and brought with him some 30 distinguished visitors from the Washington area, including Admiral C. M. Bolster, Chief of Naval Research, Commander R. C. Truax, Milton Rosen of the Naval Research Lab, and Lt. Col. Franco Fiorio of the Italian Embassy.

Southern California. About 365 diners turned out on October 22 to hear Wernher von Braun of Redstone Arsenal lecture on space travel. A November 24 dinner meeting drew 60 members to hear Prof. Leverett Davis speak on gyroscopes.

New Mexico-West Texas. A special Christmas bulletin on the Section's activities, carrying local advertising, was mailed to members and selected nonmembers on December 21.

Greer and Kearfott Join ARS

Two eastern manufacturers joined the ARS roster of Corporate Members recently. They are Greer Hydraulics, Inc., Brooklyn, N. Y., and Kearfott Co., Little Falls, N. J.

Greer, manufacturer of aviation test equipment and hydraulic components, is an 11-year old organization well known in the jet propulsion field. It is currently doing research and development work on equipment adapted to guided missiles.

Kearfott, a subsidiary of General Precision Equipment Corporation, produces mechanical, electrical, and electronic components including gyros, servos, tachometer generators, and aircraft navigational systems.

Pendray on Nationwide TV for ARS

The story of ARS was propagated throughout the country via television on November 24 when G. Edward Pendray appeared on the Bob Considine Show carried by NBC.

Dr. Pendray, a founder of the Society and a current member of the Board of Directors, was interviewed by Considine on the formation of ARS, its current operations, and its future plans.

A film supplied to NBC to ARS through the courtesy of George Benedict of North American Aviation, Inc., was run during part of the show. It illustrated the firing of a NATIV missile.

ARS to Hold Joint Session with IAS

Four papers will be presented by ARS at a Rocket Propulsion session scheduled by the Institute of the Aeronautical Sciences on Jan. 29 in New York.

The session, to be held at 2:00 p.m. in the North Ballroom of the Hotel Astor, will have Andrew G. Haley as Chairman and will be the final one of the Twenty-Second Annual IAS Meeting.

The papers are:

"Ballistics of Evaporating Droplets," by C. C. Miesse of Aerojet-General Corp.

"Gas Torch Igniter for Rocket Motors," by George N. Woodruff of Reaction Motors, Inc.

"Toxicity and Health Hazards of Rocket Propellants," by Stephen Krop, Army Chemical Center, Maryland.

"High Altitude Research with Rocket Aircraft," by Kurt Stehling of Forrestal Research Center, Princeton University.

Air Force Tests Man to 45 Gravity Force

EXPERIMENTS using rocket sleds have proved that humans can withstand acceleration or deceleration of 45 gravities with only "pressure-type" skin bruises resulting, according to Dr. A. W. Hetherington of the Air Force's Department of Human Factors of the ARDC, Baltimore.

The extremely high accelerations are exerted for only brief periods, said Dr. Hetherington at a meeting of the Northeastern New York Section of the American Rocket Society at Union College, Schenectady, N. Y., on Dec. 15. Further, results indicate that it may be possible to subject the human body to 100 or 150 gravities if certain conditions—particularly proper physical support for the body—are strictly met. If the subject is not given sufficient support an acceleration of even 5 g could produce temporary blindness and loss of hearing, said Hetherington.

Dr. Hetherington, with Colonel John Talbert and Lt. Col. George Long, USAF, both of ARDC, discussed other studies being undertaken by their department, including the prolonged existence of of mammals in sealed cabins; oxygen generation from solid materials; the heat tolerance of human beings (it has been shown that man can perform light tasks for about 6 min in a 200 F atmosphere); the effects of cosmic radiation; and the effect of subgravity on the body. Experiments concerned with the latter problem were performed using conventional aircraft in a ballistic trajectory. The subgravity effect, similar to that produced in a rapid elevator descent, was achieved for a period of 18 sec in some cases.

Book Reviews

H. S. SEIFERT, California Institute of Technology, *Associate Editor*

Conquest of the Moon, by W. von Braun, F. L. Whipple, and W. Ley; C. Ryan, Editor, The Viking Press, Inc., New York, N. Y., 1953, 126 pp. $4.50.

Reviewed by H. S. Seifert
California Institute of Technology
Jet Propulsion Laboratory

Under the auspices and editorship of *Collier's* magazine, the authors have written another speculative opus similar to their book "Across the Space Frontier." In 36,000 words and 16 beautiful illustrations, they discuss the satellite base, the assembly of a fleet of three moon ships, the orbital take-off and landing on the moon, the lunar base and its vicissitudes, and the return trip. As the embodiment of a space ship becomes more sharply focused with advancing technology, the story of the moon flight takes on added excitement and plausibility. Only the most sophisticated reader could find reasons to criticize this book, which documents its assertions with numbers and which violates no physical principles.

The authors take the position that all the elements necessary for space flight are at hand and that indeed such flight can be considered more probable than were radio communication and airplane flight only 40 years ago. They maintain that, given a strongly motivated and carefully planned effort (plus a few billion dollars), a satellite is possible in 15 years and a lunar flight in 25 years. Opposition to this point of view is felt by some guided-missile engineers, who are unable to display all the evidence behind their conclusions because of security restrictions. These engineers feel that the time scale, the technical difficulties, and the monetary cost of the project have been underestimated by perhaps an order of magnitude. They agree that in the natural course of events space flight will be attempted but that some motivation other than scholarly curiosity will be needed to accomplish it in the brief period of a decade or two. In this respect space travel differs from the development of the A-bomb, which was motivated by no less than the desire for survival.

This reviewer thoroughly enjoyed the book and felt a strong sense of identification and "escape" while reading it. It will certainly turn a profit for its publisher. However, though most of what it says is correct or at least possible, what is left unsaid is also important. For example, on page 54, the discussion as to how a ship could return to its orbital base in the event of a guidance failure on take-off glosses over a multitude of difficulties and has a certain fairy-tale atmosphere of unreality. This same attitude of dedicated optimism permeates much of the book. It is not likely, however, that such a well-written and stimulating book will in the long run do anything but good for the cause of rocketry.

V2—Der Schuss ins Weltall, by W. Dornberger, Bechtle Verlag, Esslingen, 1952, 295 pp. About $2.20.

Reviewed by A. J. Zaehringer
Thiokol Corporation, Elkton, Md.

"V2—The Shot into Space" is subtitled by Dr. Dornberger, the former military commander of wartime Peenemuende, as "The History of a Great Device." This small book is written in semitechnical language and reads much like a novel, giving an authoritative history of German rocket development with particular respect to the V2. Although the technical details of developments by now are quite well known, this book is valuable for its historical and political story. However, the main emphasis of the treatment tends to the political evolution of rocketry and the story of how a handful of enthusiastic researchers such as Oberth, von Braun, and the author brought forth the V2 as a weapon. The volume is made attractive by its interesting photographs, many of which have never before appeared in print. Since this book is in German and since few new technical details are offered, it will see little circulation in technical circles. Its main value will be in the politics of rocket evolution.

Detonation in Condensed Explosives, by J. Taylor, Oxford University Press, New York, N. Y., 1952, 196 pp. $5.

Reviewed by L. Carleton
Aerojet-General Corporation,
Azusa, Calif.

If the steam engine fathered thermodynamics, the hydrodynamic theory of detonation traces its pedigree to Nobel's inventions of dynamite and blasting gelatin. This book illustrates the way in which a science grows from its beginnings of bare experimental fact. The example commands special interest because the growth takes place under the most stringent conditions; the development of "permitted explosives," which have enough power to shatter rock masses but are slow and cool enough in reaction to avoid igniting the lethal air-methane mixtures of the coal mines, was literally a matter of life and death.

Although the book summarizes years of intensive work, both at the Nobel Division of Imperial Chemical Industries (of which Mr. Taylor is director) and elsewhere, it keeps its compactness, and the factual information is always subordinated to the main purpose: the development and use of the hydrodynamic theory. Certainly this is a refreshing change from the catch-all explosives volumes of the past generation.

The heart of the theory is the Rankine-Hugoniot equation for shock waves and the Chapman-Jouguet condition for their propagation by chemical reaction. Development of these topics is careful and thorough. The resulting equations are first applied to reactions in gaseous mixtures, where pressures are so low that the equation of state is that of a perfect gas, with no "adjustable constants." The closeness with which experimental detonation velocities are predicted verifies the theory in a thoroughly satisfying way.

The theory handles detonation in condensed media with less assurance, due to the lack of a generally applicable equation of state for the pressures of hundreds of thousands of atmospheres attained in the reaction products. Some equations contain covolumes which must be evaluated by the detonation experiments themselves; then an independent experimental check is impossible. Moreover, detonation velocity in a given explosive varies with the packaging and dimensions of the cartridge. Reaction products are complex mixtures, often heterogeneous in form, and uncertainty as to their exact nature affects not only the equation of state but also the heat of explosion. This complexity is particularly associated with commercial blasting explosives, which are moderated with various diluents to give safe temperatures and detonation velocities.

Since this book has as its practical object the improvement of these commercial explosives, the treatment herein has little to say about another interesting field of explosives application, namely, the military field. Among military explosives, high density, high detonation velocity, and high heat of explosion are at a premium. There is some feeling that in existent materials the values of these parameters have already been pushed near to their limits for conventional molecules, and that bond energies and other atomic constants simply have nothing further to give. It would be interesting to see what new ideas the methods of the book could contribute to this question.

In any case, the reader is impressed, not by the omission of a few pet topics but rather by the unusually broad coverage in so compact a book. A full development of the theory of the detonation wave is presented, together with an account of its full successes and its qualified successes and a clear indication of the gaps of knowledge remaining to be filled, as in the mechanisms of explosion reactions. This is creditable work.

Fatigue of Metals, by R. Cazaud, Philosophical Library, Inc., New York, N. Y., 1953, 344 pp. $12.50.

Reviewed by E. E. Sechler
California Institute of Technology

This book, with over 500 references, furnishes an excellent review of the efforts that have been made to determine the fatigue characteristics of metals. Obviously all of the great mass of data that

have been collected in the past half century could not be reproduced, but the author has made a sampling of these data, which are carefully designed to bring out the major parameters affecting fatigue failure. The data presented are relatively unique in that, whenever available, complete descriptions of the chemical compositions of the metal tested are presented in addition to the fatigue data.

The major emphasis of the book is on the ferrous alloys, although a limited amount of information on aluminum and magnesium alloys is also given. Since the light alloys are becoming of increasing importance in design, specific information on these materials would have been of considerable value.

For the United States reader, the use of metric units throughout may be somewhat confusing. This is no criticism of the author but is mentioned merely as a warning to such readers that they should review the conversion factors from metric to English units.

The last chapter is devoted to a review that might be called the designer's responsibility in fatigue. The importance of design techniques is best brought out by quoting the author's own comment on this subject: "Too often, culpability for fatigue failures and accidents resulting therefrom is thrown on to the metallurgists who elaborated, transformed or treated the metal, although it is in fact the designer who has transgressed in deciding the shape or dimensions of a component."

The book is very well written, and the author avoids controversy in such a controversial subject as fatigue by presenting well-authenticated experimental data from which the reader can draw his own conclusions. The original edition was dated 1948, but this translated version contains numerous references and experimental results of later date. The book will be of considerable value to both designers and research workers in the field of fatigue.

Mathematical Methods for Scientists and Engineers, by L. P. Smith, Prentice-Hall, Inc., New York, N. Y., 1953, 453 pp. $13.35.

Reviewed by R. M. Stewart
California Institute of Technology
Jet Propulsion Laboratory

This book is a collection of methods of mathematical analysis, mostly given without formal proof, which are frequently used in applied science and engineering. The author is chairman of the Department of Physics at Cornell University and was formerly associated with the Atomic Energy Program.

Many of the chapters can be considered to be only brief reviews. The outstanding example of this fact is the chapter on probability, to which 22 pages are devoted. The recent book by Hildebrand ("Methods of Applied Mathematics," same publisher!) covers most of the material which might recommend this book as an addition to a personal library (with the exception of the chapters on complex variable); the price is less than half that of the present book and, in this reviewer's opinion, should appeal to a wider audience. The chapter titles of Smith's book are as follows: (1) Elements of Function Theory; (2) Differential Calculus; (3) Integral Calculus; (4) Space Geometry; (5) Line, Surface, and Multiple Integrals; (6) Theory of Functions of a Complex Variable; (7) Residues and Complex Integration; (8) Representation of Functions by Infinite Series of Functions; (9) Application of Functions of a Complex Variable to Potential and Flow Problems; (10) Algebra of Linear Equations, Transformations and Quadratic Forms; (11) Vector and Tensor Analysis; (12) Orthonormal Function Systems; (13) Orthonormal Functions with a Continuous Spectrum; (14) Integral Equations; (15) Variational Methods; and (16) Elements of Probability Theory.

Measurement Techniques in Mechanical Engineering, by R. J. Sweeney, John Wiley & Sons, Inc., New York, N. Y., 1953, 309 pp., $5.50.

Reviewed by J. F. Manildi
University of California at Los Angeles

The author has presented, in one book, a comprehensive discussion of measurement techniques currently used in practice. A review of basic measurement theory is presented. In many cases the underlying mathematical theory for the instruments discussed is presented. Standard electrical measurement, dynamometers, pressure and temperature measurement, fluid flow, calorimetry, chemical analyzers, and automatic controls are covered.

The section on electrical measurement and fluid measurement offers nothing new. The sections on pressure and temperature measurement are notable in that the author discusses the very pertinent subject of errors in measurement. The short section on automatic control is primarily a qualitative discussion of various control devices in use.

Specialized instrumentation, such as accelerometers and many instruments used in aircraft and guided-missile application, is understandably not discussed.

The book is a useful reference text for laboratory courses and is also of practical value to the practicing engineer because of its broad coverage.

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Technical Literature Digest

M. H. SMITH and M. H. FISHER

The James Forrestal Research Center, Princeton University, Princeton, N. J.

Jet Propulsion Engines

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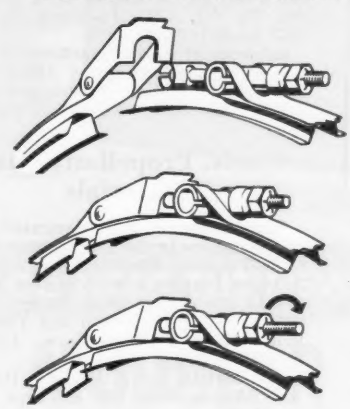

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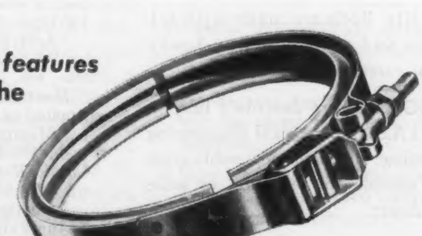

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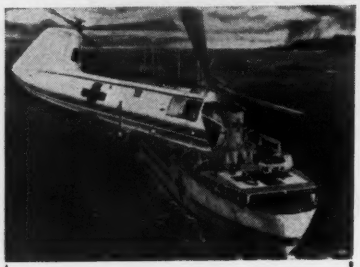

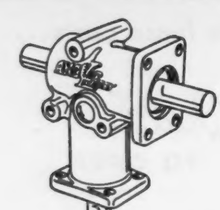

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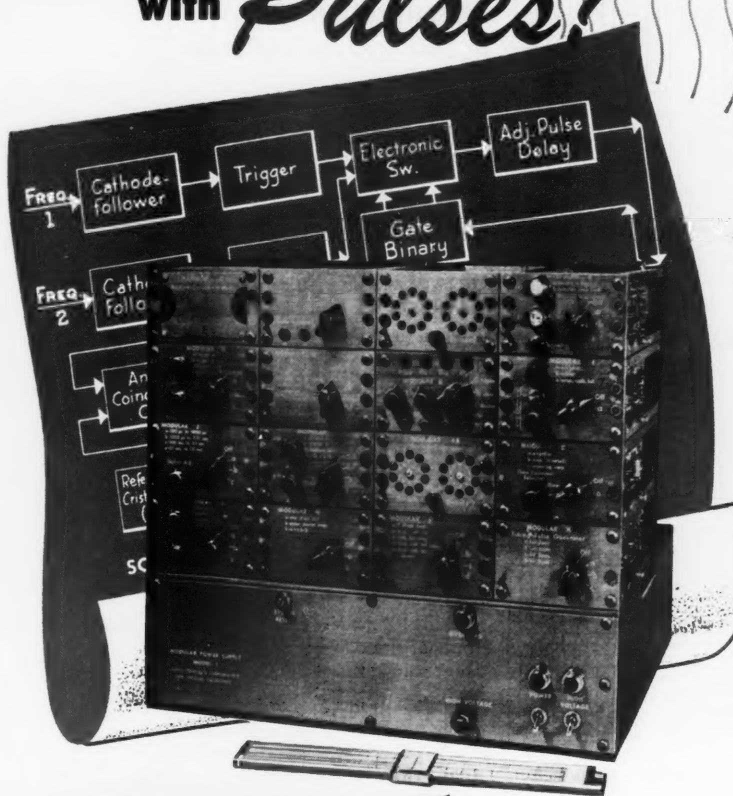

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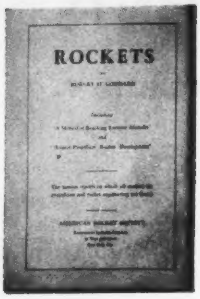

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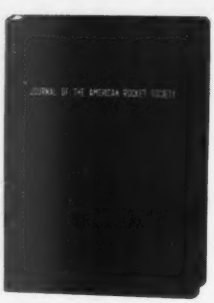

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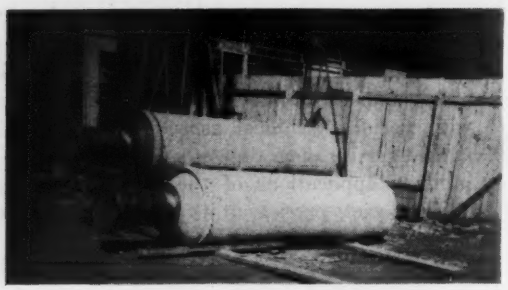

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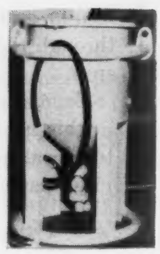

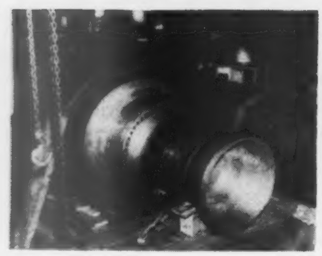

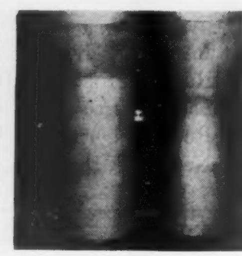

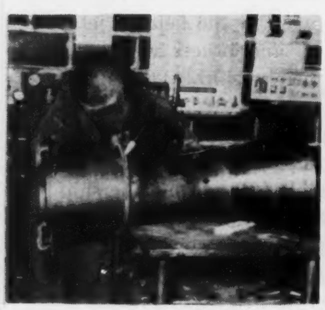

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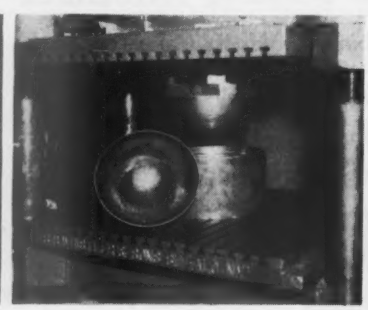

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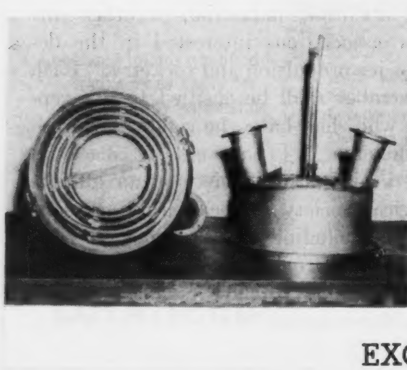

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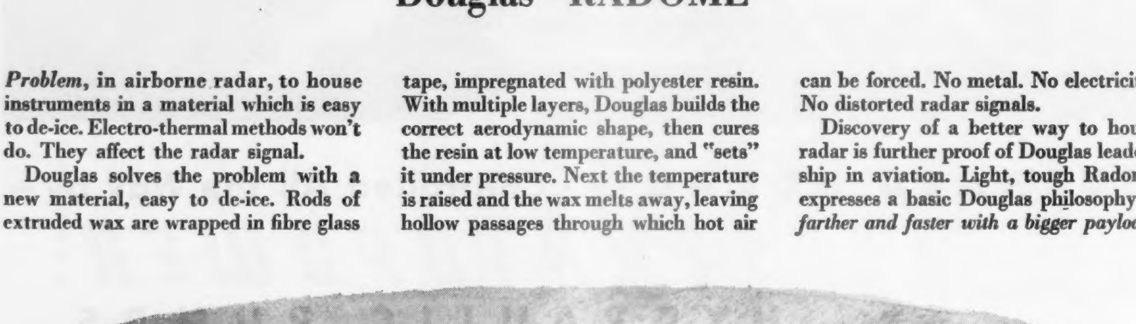

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A ROCKET DRIVE FOR LONG-RANGE BOMBERS

By Eugen Sänger and Irene Bredt

The only English translation available to the public of this report (condensed) on the problems of long-range military rocket aircraft, originally published in German. Catalogs new problems and outlines solutions to the more important ones; discusses launching and climb, gliding flight and landing projection of bombs, types of attack, and the line of development of the rocket bomber. Will serve for years to come as a storehouse of vital concepts for the serious student of rocket science. Numerous maps, charts.

Published 1952 | 175 pages | $3.95

MICROMETEOROLOGY

By O. G. Sutton

The earlier chapters, in this book, treat of the basic physics and mathematics required for the study of processes taking place near a boundary such as the surface of the earth. The later chapters show how the fundamental concepts are applied to yield solutions which, although necessarily approximate, have important consequences in practical problems of hydrology, atmospheric pollution, and agricultural meteorology.

Published 1953 | 323 pages | $8.50

RAKETENANTRIEBE

By J. Stemmer

This book (in German) was written as an easily readable introduction into the field of rockets and space travel on a not-too-technical basis. A large section of the book is dedicated to an encyclopedic description of rockets and rocket-propelled airplanes that were actually built by the major contestants during World War II, and contains a wealth of details on construction and performance from the "Katiushka" of the eastern front to the V-2 missiles and the "Bat" in the west.

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EXTERIOR BALLISTICS

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The authors were charged with study and development of the science of ballistics in the experimental work performed during World War II. Here is presented the product of their work.

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FLYING SAUCERS

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With humor, imagination, and common sense, Dr. Menzel describes all types of apparitions. He reviews the flying-saucer stories that started and perpetuated all the mystery. He explains, simply and clearly, how interactions between light and atmospheric conditions cause the remarkable variety of phenomena—from the aurora borealis and sundogs to radar mirages—that occur in the sky. He concludes with a serious study of the possibilities of interplanetary travel and an evaluation of flying saucers in some future era.

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NUCLEAR PHYSICS

By W. Heisenberg

This work begins with a short and interesting history of the views about atoms in antiquity and also of the development of atomic theory till the close of the nineteenth century. The main subjects in the book, which includes radioactivity, are binding energy of nuclei, nuclear structure, artificially induced nuclear transmutations and also the methods of observation and of producing nuclear transmutations.

1953 | 225 pages | $4.75

COMPLEX VARIABLE THEORY AND TRANSFORM CALCULUS

By N. W. McLachlan

This is the second edition of the book first published in 1939 under the title "Complex Variable and Operational Calculus With Technical Applications." Certain of the old sections have been removed to make way for more important subject matter; e.g. repeated impulses, Fourier transforms, and frequency spectra.

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DISCONTINUOUS AUTOMATIC CONTROL

By Irmgard Flügge-Lotz

Discontinuous automatic control systems are assuming increasing importance in engineering applications, especially in devices such as missiles, which must be expendable. Dr. Flügge-Lotz, who pioneered in this field in Germany and now continues her research at Stanford University, has shown how tedious computations can be replaced by graphical solutions.

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HIGH ALTITUDE ROCKET RESEARCH

By Homer E. Newell, Jr.

This book describes the use of rockets for studying the ionosphere, cosmic rays, Earth's magnetic field, atmosphere, and solar radiation. Research and instrumentation techniques as well as experimental difficulties peculiar to rockets are surveyed. Dr. Newell describes the WAC Corporal, V-2, Aerobee, and Viking rockets, and lists all the upper-air research firings conducted by United States agencies from early 1946 to December 31, 1952.

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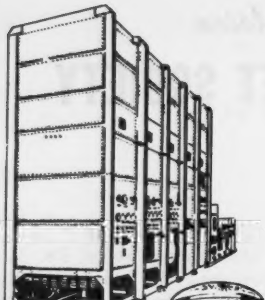

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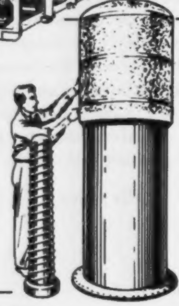

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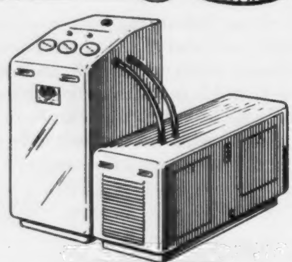

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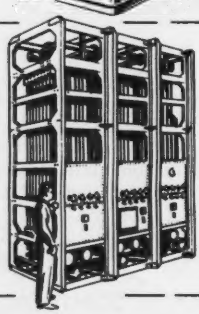

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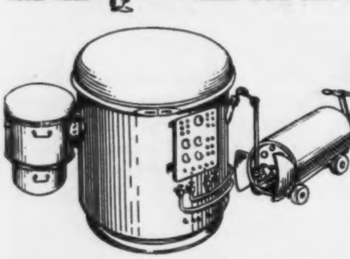

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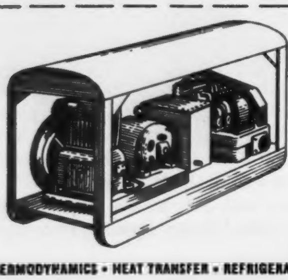

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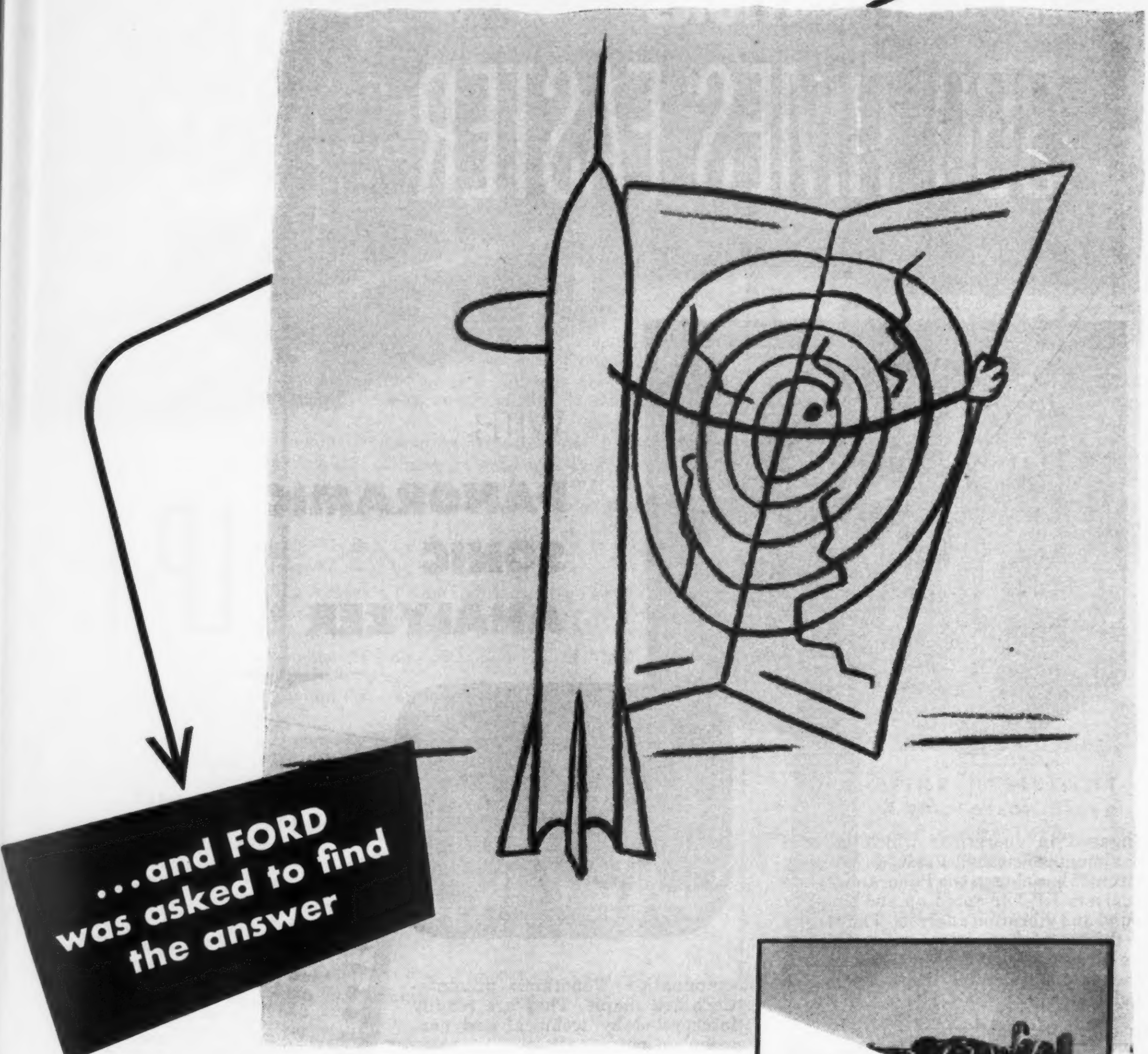

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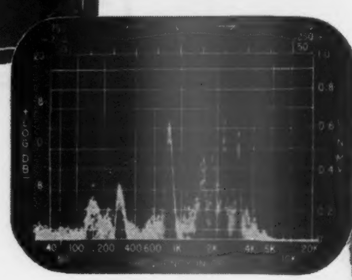

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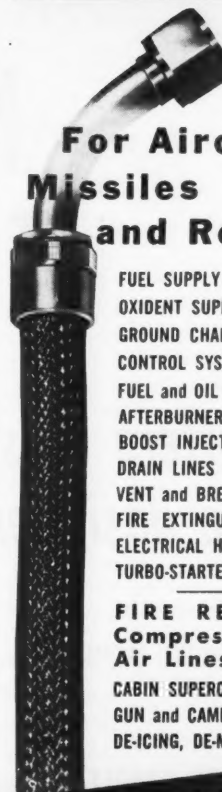

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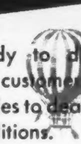

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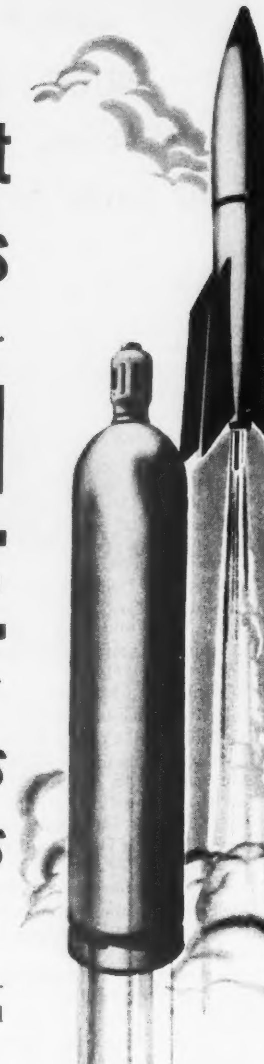

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